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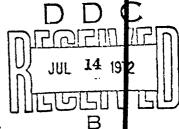
VOLUME I OF II

FLIGHT TEST AND WIND TUNNEL PERFORMANCE CHARACTERISTICS OF THE X-24A LIFTING BODY

LAWRENCE G. ASH Captain, USAF Aerospace Research Engineer

TECHNOLOGY DOCUMENT NO. 71-8

JUNE 1972



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FOREWORD

This technology document presents the performance characteristics of the X-24A lifting body obtained from flight data and compares them with wind tunnel predictions. The X-24A flight test program began 17 April 1969 and was completed on 4 June 1971. References 1 through 8 are related documents reporting the flight test results from the X-24A program.

The author wishes to acknowledge the efforts of Mr. David F. Richardson for developing the computer program used to calculate performance parameters from flight data (appendix IV) and Mr. Christopher J. Nagy who developed the X-24A position error correction (appendix III). The author also wishes to acknowledge Mr. Robert G. Hoey for his study on tip fin flow separation (appendix V) and for his advice and assistance in writing this report.

The participation of AFFTC personnel in this program was authorized by Project Directive 69-38. The assigned Program Structure was 680A.

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ABSTRACT

The objectives of determining the performance characteristics of the X-24A lifting body from flight test data and correlating these results with wind tunnel predictions were successfully accomplished. Lift and drag characteristics were computed from onboard measured accelerations and flight conditions while the vehicle was in gliding flight. Performance data were obtained over Mach number and angle of attack ranges of 0.26 to 1.5 and 1.5 to 19.6 degrees, respectively. Discrepancies were apparent between different wind tunnel predictions. However, flight test data generally exhibited lower lift and slightly lower drag than wind tunnel predictions with the result that flight test and wind tunnel L/D values were in reasonably good agreement. Effects of Mach number, vehicle control surface configuration, and landing gear deployment on performance parameters were determined. Degradation of flight performance at low subsonic Mach numbers was attributed to tip fin flow scparation. In Volume II of this report, supersonic performance comparisons are made between the PRIME and X-24A vehicles, both of which were of the same aerodynamic configuration (SV-5). A significant degradation in subsonic performance due to simulated ablative surface roughness was obtained in full scale wind tunnel tests. Flight testing of simulated ablatives on the X-24A was not pursued; however, the estimated effects of the ablatives are discussed in Volume II.

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These CONFIDENTIAL appendicies will be published separately as Volume II to this report.

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list of abbreviations and symbols

Item	Definition	Units
$\mathfrak{a}_{\mathrm{x_b}}$	<pre>acceleration along body x-axis (longitudinal)</pre>	g's
a _z _b	acceleration along body z-axis (normal)	g's
b	reference span (10 ft)	ft
ē	reference length (23 ft)	ft
c _e	chord force coefficient	dimensionless
c _{c5e}	∂C _C /38e	per rad, per deg
c_{D}	vehicle total drag coefficient	dimen; ionless
c_{D_0}	<pre>zero lift drag coefficient (trimmed drag at zero lift)</pre>	dimensionless
C_l	vehicle lift coefficient	dimensionless
$c_{\mathtt{L_{i.c}}}$	<pre>vehicle lift coefficient based on indi- cated airspeed (corrected for instrument error)</pre>	dimensionless
$c_{\mathbf{L}_{lpha}}$	∂C _L / ∂ α	per deg
Ci	rolling moment coefficient	dimensionless
C _i	∂ C _x /∂β	per dou
c , sa _u	aC _€ /a8a _U	per deg
C^{m}	pitching moment coefficient	dimensionless
$c_{m_{\mathbf{q}}}$	$\delta C_{\rm m} / \delta \left(\frac{{\rm q} \bar{c}}{2 {\rm V}_{\rm L}} \right)$	per rad
Cm ₆ e	∂C _m /∂∂ _e	per rad, per deg
c_N	normal force coefficient	dimensionless
$c_{N_{\alpha}}$	∂C _N /∂a	per deg
c _{N se}	ac _N /33e	per rad, per deg
c_n	yawing moment coefficient	dimensionless
$c_{n_{\vec{\beta}}}$	∂C _n /∂∂	per deg
$c_{n_{\delta}a_U}$	əCn∕əδa _U	per deg
cg	center of gravity	pct c
H	pressure altitude	ft
нм	hinge moment	in1b
lH _{PC}	altitude position error correction	ft
L/D	ratio of lift to drag	dimensionless
L/D _{max}	maximum ratio of lift to drag	dimensionless
M	flight or free stream Mach number	dimensionless

<u>ltem</u>	Definition	Units
Mic	<pre>indicated Mach number corrected for instrument error (not corrected for position error)</pre>	dimensionless
$^{\Lambda M}$ pc	Mach correction for position error	dimensionless
NASA	National Aeronauties and Space Administra- tion	~ <u>-</u> -
OD	outer diameter	in.
þ	roll rate	deg per sec
$\dot{ m p}$	roll acceleration	dag per sec ²
P_a	atmospheric or ambient pressure	рз f
PCM	pulse code modulation	
$q^{q} \wedge$	static pressure source position error	psf
^Pp/qcic	position error pressure coefficient	dimensionless
P_{S}	measured static pressure	psf
q	pitch rate	deg per sec
q	pitch acceleration	deg per sec ²
q _c	differential pressure	psf
qcic	indicated differential pressure corrected for instrument error	psf
r	yaw rate	deg per sec
r	yaw acceleration	deg per sec ²
R_{Q}	Reynolds number	dimensionless
S	reference area (52 ft ²)	rt ²
SAS	stability augmentation system	
'n	thrust	1b
v_e	equivalent airspeed (= V _t $\sqrt{5}$)	kt
$^{4}\mathrm{v}_{\mathrm{PG}}$	airspeed position error correction	kt
v_t	true airspeed	kt
W	airplane gross weight	lb
1	true angle of attack	deg
"boom	vane angle of attack with respect to the nose boom	deg
$^{\Lambda a}$ bb	correction to angle of attack for boom bending	deg
i	indicated angle of attack	deg
$^{\lambda_{\mathbf{q}}}$	correction to angle of attack for pitch rate	gr À

Itom	Definition	Units
"T	true angle of attack as established by the wind tunnel	deg
2.4	correction to angle of attack for upwash	deg
3	sideslip angle	deg, rad
śa _U	upper flap aileron deflection	deg, rad
ĉe .	elevator deflection	deg
°e _L	lower flap defrection in pitch	ueg
δeU	upper flap deflection in pitch	deg
$^5 extsf{L}_{ extsf{B}}$	lower flap bias position	deg
${\rm SR}_{\rm B}$	rudder bias position	deg
$5 { m U}_{ m B}$	upper flap bias position	deg
δW	wedge angle ($\delta w = \delta L_{\rm B} + \delta e_{\rm D}$ or $\delta e_{\rm L} + \delta U_{\rm B}$)	deg
S	relative air density or air density ratio (s = s/r _{SL})	dimensionless





INTRODUCTION

The X-24A lifting body flight test program was conducted at the Air Force Flight Test Center between April 1969 and June 1971. In addition to proving the unpowered landing capability of this class of medium lift-to-drag ratio (L/D) reentry vehicle, extensive flight data were obtained to define the handling qualities, stability, control, and performance characteristics of the X-24A configuration through the subsonic, transonic, and supersonic Mach number (M) regions up to M=1.60.

Design predictions of the flight characteristics of the X-24A lifting body were based on theoretical calculations and wind tunnel data obtained from small scale models. Verifying the accuracy of these data and investigating the effect of any inaccuracies on vehicle performance were major objectives of the flight test program. To accomplish these objectives, performance data were obtained from maneuvers on a flightby-flight basis for the entire test program. This report presents the performance characteristics obtained from the 28 X-24A flights. Data were obtained for 7 fixed upper flap bias configurations and 3 rudder bias configurations, over a Mach number range of M = 0.26 to 1.50 and at angles of attack (a) from 1.5 degrees to 19.6 degrees. Flight Reynolds numbers (Re) ranged from 12 x 100 to 65 x 106, based on the vehicle reference length of 23.0 feet. Some data were also obtained with the lower flaps at zero degrees while controlling with the upper flaps in both the landing gear up and landing gear down configurations. Flight results were compared with full scale and small scale wind tunnel predictions. Effects of Mach number, wedge angle, rudder bias, tip fin flow separation, and landing gear deployment on flight performance parameters are analyzed and discussed.

In appendices VI and VII to this report (published separately), hypersonic flight data from the subscale PRIME² test program were compared to the supersonic X-24A flight performance data. Also included is a study which summarized the results of several tests 'including a full scale X-24A wind tunnel test) dealing with the effects of ablatives on the low speed performance characteristics of medium L/D vehicles. These effects were applied to X-24A flight test performance data in an attempt to predict the subsonic performance characteristics of an ablated X-24A.

DESCRIPTION OF THE VEHICLE

The overall shape and dimensions of the X-24A are shown in figure 1. The vehicle was wedge-shaped in planform with a flat bottom and the top a curved airfoil surface with three vertical fins. All control surfaces were located at the aft end of the vehicle and consisted of two upper and two lower flaps and two upper and two lower rudders on the outboard fins. Pilot and stability augmentation system (SAS) inputs in pitch and roll were transmitted mechanically to the lower flap actuators. When either lower flap reached the fully closed position (zero do rees), pitch and roll inputs were transferred through a clapper and namism to the corresponding upper flap. Pilot rudder pedal and yaw SAS inputs caused motion of the upper rudders only as yaw control surfaces. Both pairs of upper and lower rudder surfaces could be deflected symmetrically (biased). Also, both pairs of upper and lower flaps could be

²PRIME (Precision Recovery Including Maneuvering Entry), an unmanned subscale vehicle of the same configuration as the X-24A (SV-5) which was boosted to arbital speeds on an Atlas booster.

brased within a range of positions. The flaps and xudder surfaces could be brased outward from their closed positions to increase the stability at trans. ic and supersonic speeds. The flap and rudder brase features were drived by slowly moving trim motors which were controlled by switches in the property control surface brased designations and sign conventions are she in figure 6. A complete description of the X-24A flight control system is presented in reference 1.

Most flying was accomplished with the upper flaps fixed while controlling with the lower flaps. However, two landing approaches were accomplished with the lower flaps at zero degrees, with pitch and roll control on the upper flaps. Upper flap control was also used for most of the landings after landing gear deployment. The vehicle configuration that evolved for flight at Mach numbers greater than 0.50 is shown in figure 2. The standard approach and landing configuration is shown in figure 3. A complete description of the X-24A vehicle is presented in reference 2.

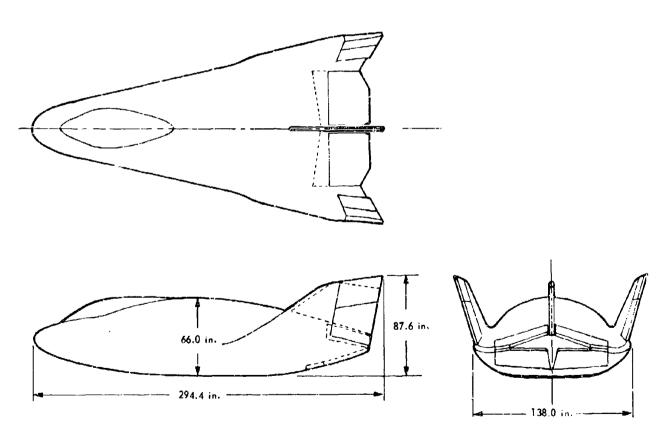


Figure 1 Three-View Drawing of the X-24A

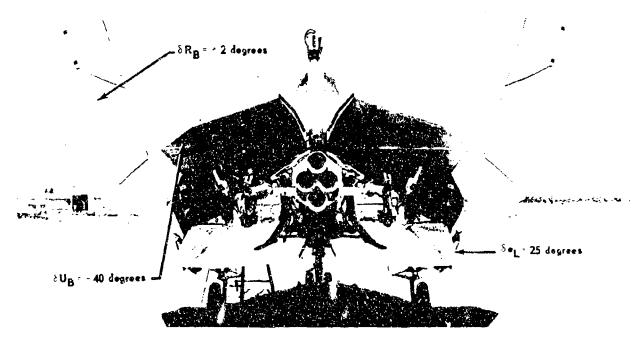


Figure 2 Transenic and Supersenic Configuration

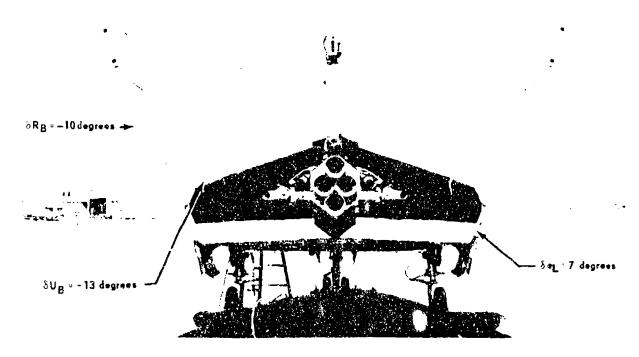


Figure 3 Subsonic Configuration

TEST METHOD

The X-24A flight program consisted of 10 glide flights and 18 powered flights using an XLR-11 rocket engine. In powered flight, the vehicle was flown to a predetermined Mach number and altitude at w.ich time the rocket engine was shut down and the vehicle glided to an unpowered landing. A complete description of flight profiles, procedures, and objectives is presented in reference 2.

All performance data maneuvers were performed with the vehicle in gliding flight. No attempt was made to obtain lift and drag data while the rocket engine was running or during propellant jettison.

Test Performance Maneuvers

A pushover-pullup maneuver was used to obtain most of the performance flight data. A typical flight maneuver is shown in figure 4. The pilot steadily decreased angle of attack to about 3 degrees, pulled up to 14 degrees, and then returned to the original angle of attack. The pilot tried to perform the task slowly enough to avoid large pitch accelerations, but fast enough to keep Mach number somewhat constant during the maneuver. Trading off these two parameters resulted in an average Mach number change of 0.055 during subsonic performance maneuvers and 0.1 to 0.27 during supersonic maneuvers. Besides the planned performance maneuvers, additional data were obtained from any substantial a excursion where the pitch rate was fairly constant.

Instrumentation

Accelerations were measured by sensitive accelerometers placed close to the vehicle's center of gravity. Angle of attack, angle of sideship, static pressure (Pg), and total pressure were measured by a standard NASA pitot static tube on a 6.2-foot nose boom. All parameters were telemetered to ground stations by means of a pulse code modulation (PCM) data system and recorded on magnetic tape at the ground station.

Data Analysis

The raw data were processed through a NASA data reduction follow-on computer program. This program made all the necessary corrections to the data including an upwash correction to angle of attack and a position error correction to measured static pressure. A detailed discussion of how these two corrections were developed is in appendices II and III.

For each maneuver, data were selected at points where pitch acceleration (g) was approximately zero. This was done in an attempt to obtain more consistent data at trimmed flight conditions. Figure 5 compares data selected at half second intervals of a maneuver, to data of the same maneuver selected at joints of zero pitch acceleration. The data selected at zero pitch acceleration was more consistent and showed a smaller scatter band.

An 1BM 1620 digital computer program was used to make additional corrections to the flight data and compute performance characteristics. Corrections were made to the accelerometers for misalignment and displacement from the cest cg. Other corrections were made to the trim elevator

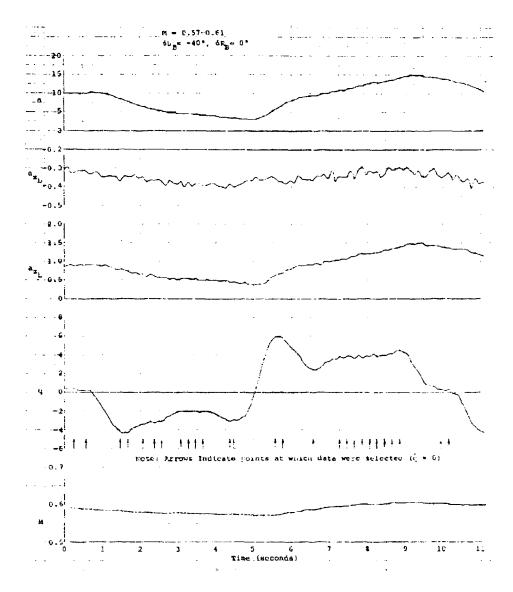
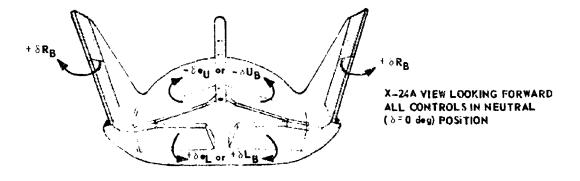


Figure 4 Time History of a Pushover-Pullup Maneuver

position for pitch rate and cg variation from a reference wind tunnel cg of 57 percent. The body axis force coefficients were computed from the corrected accelerations, calculated gross weight (reference 3), and dynamic pressure. Lift coefficient (CL), drag coefficient (CD) and L/D were then computed by rotating the body axis coefficients to the stability axis using true angle of attack. The body and stability axis coordinate systems showing the positive direction of forces, moments, and angles are presented in figure 6. A detailed description of the computer program along with a list of equations, a program listout, and a sample output are in appendix IV.

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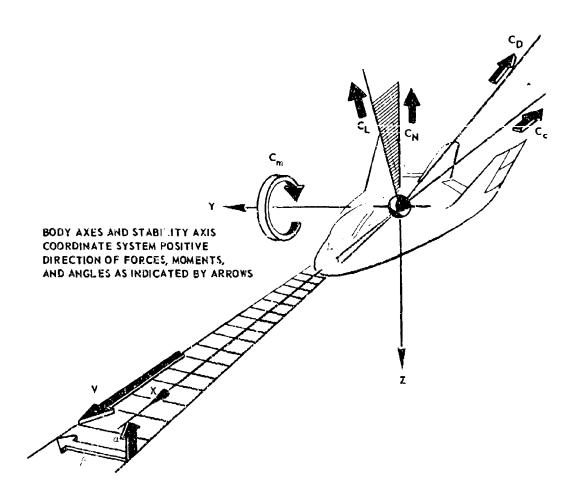


Figure 6 Control Surface Bias Designations, Axis Systems, and Sign Conventions

WIND TUNNEL TESTS

Extensive wind tunnel tests were performed on the full scale X-24A (at M = 0.2) and an 8 percent steel model of the aircraft prior to the flight test program. A wide range of vehicle control surface configurations were tested for a Mach number and angle of attack range which oncompassed the X-24A flight test envelope. Table I presents a summary of the wind tunnel tests that were used for comparison with flight data in this report. Since the performance data were greatly affected by control surface configuration and Mach number, only wind tunnel data which corresponded to the configuration and flight conditions of a particular test maneuver were used for comparison. There was an exception in the case of the full scale tunnel data which were limited to 0.2 Mach number. These data were only compared to flight data below M = 0.5 in the approach and gear down landing configurations. All other flight data were compared to tunnel tests performed on eight percent models.

Most wind tunnel data taken with the vehicle in the transonic configuration, (upper flap bias position, $\delta U_{\rm B} = -30$ to -40 degrees with rudder bias position, $\delta R_{\rm B} = 0$ degrees), were obtained from the Langley 8-foot wind tunnel. Most tunnel data taken at the low subsonic and approach configuration (sUB = -10 to +30 degrees, $\delta R_{\rm B} = -10$ degrees), were obtained from the Langley 7- x 10-foot wind tunnel.

Additional wind tunnel tests were performed later in the flight test program at the Cornell Aeronautical Laboratories using an eight percent fiberglass model. The main purpose of these tests was to obtain additional stability and control information, and only a small amount of performance data was obtained.



Table I SUMMARY OF WIND TUNNEL TESTS

Wind			, ,	Upper	Lower		
		Mach			Setting	Setting	R × 10 ⁻⁶
(!txft)	Date	No.	(deg)	(deg)	(deg)	(deg)	
		HASA	- Langley	(8 pct X-24A	model)		
8×8	Apr 1965	0.60	0 to 20	-40	20	0 -	5 - 86
	Oct 1965	0.80	0 to 24	1	20,30	l l	6.96
		0.90	0 t.o 24	1	20,25,30,	1	7. 2
	•					 	
	1	0.95	0 to 18			ł	
		1.00				1 1	
	1	1.20	<u> </u>			1 1	
	1	0.80	0 to 24	- 35	20		
	ł	0.90	0 to 24	_	20,25		
	i	0.95	0 to 18		20,25]	
	ļ	0.40	0 to 24	-30	0,10,20	1 1	
		0.60		ļ	10,15,20	1	
		0.80		ge Setting (deg) (10,15,20	<u></u>	
7×10	Jun 1968	0.50	-4 to 20		С	-10	5.41
	Jul 1968	0.50	-4 to 22	- 30	20	!	5.41
	ł	0.60	1	- 30	20]	6.42
	ļ	0.60	1	-25	10,15,20		6.42
	ì	0.70	1	-25	10,15,20	1	7.18
	1	0.50	1	-20	5,10,15,20	<u>,</u>	5.41
İ	İ	0,60	j	Ì		į	6.42
Į		0.70]	1		i	7.18
	1	0.50	1	-15	5,10,15	7	5.41
		North An	erican Avı	ation (8 pct	X-24A model	1)	
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	Date Nach Range Stiting St		l ~				
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	1	0.00	_			4	1
		0,20		-20			
		0.20	7	-20**	10,20	7	
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*Gcar down **With simulated ablatives

COMPARISON OF FLIGHT DATA TO WIND TUNNEL RESULTS

Performance flight test data are presented in figures 1 through 30, appendix I. Plots of $C_{\rm L}$ and $C_{\rm D}$ versus angle of attack, and L/D and $C_{\rm D}$ versus CL are shown for all performance maneuvers flown and analyzed. All flight test data points were trimmed data corrected to a 57 percent reference og for comparison with tunnel data. Curves were faired through the flight data points and are presented as dashed lines in the figures. The plots are sequenced in order of decreasing upper flap bias configuration, ending with data from the closed up approach configurations and gear down data. For each upper flap configuration, a set of performance data is presented at each individual Mach number for which flight test data were obtained and placed in order of increasing Mach number. A complete log of all flight test performance maneuvers is presented in table I, appendix I. For each maneuver, flight data were compared to all available wind tunnel trim points (zero pitching moment coefficient, for that flap setting) obtained at the corresponding Mach number and vehicle configuration. Where two or more trim wind tunnel points were available at one upper flap setting, a curve was faired through them to represent a trimmed wind tunnel curve at that upper flap configuration.

The Cornell wind tunnel trim points for the configuration in which $\delta U_{\rm B} = -40$ degrees and $\delta c_{\rm L} = 25$ degrees generally exhibited much higher trim angles of attack than both the Langley 8-foot tunnel and the flight data. This is suspected to be the result of a small error in the tip fin dihedral for the fiberglass model. It has been assumed that this discrepancy affected only the pitching moment data. The Cornell data points used for comparison with trimmed flight data were therefore selected at the flight determined trim angle of attack for that flap configuration.

During the flight test program there were no planned attempts to gather data at different Reynolds numbers. In some cases, however, two or more performance maneuvers were accomplished at the same flight conditions and vehicle configuration, but with different Reynolds numbers. In the flight data presented in figures 3, 5, 9, 11, 17, 27, and 28, appendix I, half solid symbols separate the maneuver which differs most in Reynolds number from the other maneuvers at each flight condition. Five different vehicle configurations and Mach numbers ranging from 0.5 to 0.9 are represented in these maneuvers. Reynolds numbers are referenced to the 23-foot reference length of the X-24A. Table II presents wind tunnel and flight Reynolds numbers for the entire flight Mach number range.

There were no variations in the flight test performance parameters which could be attributed to heynolds number effects within the range of Reynolds numbers flown. Since the flight values of Reynolds number were always significantly larger than corresponding wind tunnel values, no conclusions could be reached as to the influence of Reynolds number on the performance discrepancies observed between flight test and wind tunnel results.

Table 11
FLIGHT TEST AND WIND TUNNEL REYNOLDS NUMBER COMPARISON

			Reyn	olds Number	× 10 ⁻⁶	
! [nel Tests		Flight	
Mach No.	Langley 8x8 ft	Langley 7x10 ft	Cornell Aero Lab	Ames Full Scale	Transonic Configuration	Subsonic Configuration
0.20				29.4		
0.35		4.32				44. δ to 54.2
0.40	4.25					
0.50		5.41	7.50		29.9 to 40.5	34.3 to 64.9
0.55					36.9, 39.6	27.5
0.60	5.83	6.42			27.2 to 34.9	35.1
0.70		7.18			18.2 to 42.1	32.3, 38.1
0.80	6.96	7.62	4.50		15.1 to 38.7	
0.90	7.32				18.6 to 30.5	
0.95	7.47		4.00			
0.97					17.3, 19.4	
1.00	5.68				14.9, 20.1	
1.10			3.75		12.5, 14.9	
1.20	5.81				12.5 to 15.6	
1.30			3.25		12.5 to 16.6	
1.40					16.6, 17.5	
1.50					17.5	

All supersonic performance flight data should be viewed with respect to the highly transient conditions under which they were obtained. A very limited amount of data was extracted from five different maneuvers during four different flights. All maneuvers were performed during periods of rapid deceleration with the aircraft undergoing AM changes of 0.10 to 0.27. In addition, this rapid deceleration was in excess of the measuring range of the sensitive longitudinal acceleranter (±0.5 g's). A less sensitive (±2.0 g's) accelerometer had to be used to compute axial chord force. Therefore, general trends of supersonic data should be well represented, but definite distinctions between data at different supersonic Mach numbers could not be determined as accurately as in the subsonic and transonic Mach number regions.

LIFT COEFFICIENT

Lift coefficient data shown in figures 1 through 30, appendix 1, were in general agreement with wind tunnel predictions at low angles of attack (two to six degrees), but dropped below predictions at higher angles of attack. Figure 7, which presents data at $\nu U_B = -40$ degrees and $\nu R_B = 0$ degrees at 0.8 Mach number, is a good example of this effect. Flight values of trimmed lift curve slope (C_{L_1}) ranged from 12 percent above to 20 percent below trimmed wind tunnel values obtained from the Langley 8-foot and 7- x 10-foot wind tunnels for all Mach numbers and vehicle configurations. Cornell wind tunnel values of lift coefficient showed excellent agreement with flight data when compared at flight trim angles of attack. The one trim lift curve obtained from the Cornell tests in the $\nu U_B = -13$ degrees approach configuration was in excellent agreement with flight data (figure 27, appendix I).

Supersonic flight lift data are presented in figures 7 and 8, appendix 1, for -40 degrees upper flap bias with 0 degrees and +2 degrees rudder bias, respectively. Supersonic flight was not performed with upper flap bias settings other than -40 degrees. Data at zero degrees fR_B from M = 1.1 to 1.2 compared well with the Langley 8-foot wind tunnel curve at M = 1.2. Flight data at +2 degrees δR_B from M = 1.4 to 1.5 showed good agreement with the North American Aviation 7- x 7-foot wind tunnel curve at M = 1.4 and +10 degrees rudder bias (figure 8, appendix 1).

Full scale wind tunnel $C_{\rm L_a}$'s were in good agreement with flight slopes in the two approach configurations. However, wind tunnel values of $C_{\rm L}$ for any particular angle of attack were generally higher throughout the x range (figures 27 and 28, appendix I).

Gear down flight data were obtained at two different control surface configurations and two mose gear door configurations. Plight data are presented in figur—29 and 30, appendix 1, along with full scale wind tunnel gear down claves. The flight data were scattered due to the different configurations, probable ground effects, and large control motions associated with the actual landings. Both sets of flight data have the same lift curve slope which is greater than the full scale wind tunnel $C_{\rm L_1}$.

³The nose gear duor was originally perpendicular to the body X-reference axis and was replaced prior to flight 9 by one canted at a 45-degree angle. Full scale wind tunnel tests were performed with the perpendicular nose gear door.

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DRAG COEFFICIENT

Prag coefficient data are plotted versus angle of attack and also as drag polars ($C_{\rm D}$ vs $C_{\rm L}$) in figures 1 through 30, appendix 1. Flight data showed slightly lower drag than the Langley eight-foot wind tunnel curves for the transonic vehicle configurations ($\delta U_{\rm B}$ = -40, -35, and -30 degrees with $\delta R_{\rm B}$ = 0 degrees) at all Mach numbers. Figure 7, which presents data in the $\delta U_{\rm B}$ = -40 degrees and $\delta R_{\rm B}$ = 0 degrees configuration at 0.8 Mach number, is a good example of this trend.

Supersonic flight test drag data at 0 degrees rudder bias and -40 degrees upper flap bias (figure 7, appendix I) were lower than Langley 8-foot wind tunnel data taken at M = 1.2. Flight drag data versus angle of attack data at +2 degrees δR_B (figure 8, appendix I) within a Mach number range of 1.3 to 1.5 showed good agreement with the North American Aviation 7- x 7-foot wind tunnel curve at M = 1.4 and +10 degrees rudder bias. Drag polar data, however, were not as consistent with the wind tunnel curve and exhibited a much different drag polar curvature (drag due to lift). The +2 degrees δR_B data were less scattered than the zero degree δR_B flight data. However, there was insufficent data to draw any conclusions as to the performance effects of positive rudder deflection.

The flight Grag data obtained at the smaller upper flap configurations ($^{3}\mathrm{U_B}$ = -8 to -23 degrees) were compared to Langley 7- x 10-foot wind tunnel curves and showed excellent agreement (figures 21 through 28, appendix I).

Cornell wind tunnel drag values obtained with the $\pm U_{\rm B}/\pm e_{\rm L}\approx -40/25$ degrees configuration and selected at flight trim angles of attack were in good agreement with flight data (figures 1 to 6, appendix 1). The trimmed Cornell drag and drag polar curves obtained with the $\pm U_{\rm B}=-13$ degrees and $\pm R_{\rm B}=-10$ degrees configuration also agreed well with flight data at low $C_{\rm L}$'s, but exhibited higher values of $C_{\rm D}$ than did the flight data at $C_{\rm L}$'s above 0.3 (figure 27, appendix 1).

Full scale wind tunnel drag versus angle of attack data were in excellent agreement with flight test data in the two approach configurations. Drag polars from the full scale tunnel form a lower boundary of the flight data scatter with the Langley 7- x 10-foot wind tunnel data forming the upper boundary. All of the flight gear down drag data were below full scale wind tunnel predictions.

LIFT-TO-DRAG RATIO

Lift-to-drag ratio flight data plotted versus lift coefficient are compared to wind tunnel data in figures 1 through 30, appendix 1. Flight data were generally in good agreement or slightly above the Langley 8-foot and 7- x 10-foot wind tunnel data. Due to stability boundaries, many of the flight maneuvers in the transonic configurations and at high mach numbers were not taken to an angle of attack high enough to reach maximum L/D. In most cases where flight maneuvers were taken to the maximum L/D, the values of maximum L/D and corresponding lift coefficient compared well with available trim wind tunnel curves.

Supersonic L/D flight data at 0 degrees rudder bias (figure 7, appendix 1) in the M=1.1 to 1.3 Mach number range were in good agreement with the Langley 8-foot wind tunnel curve at M=1.2. Flight L/D data

at +2 degrees δR_B in the M = 1.3 to 1.5 range show good agreement with the North American Aviation 7- x 7-foot wind tunnel curve at M = 1.4 and δR_B = +10 degrees (figure 8, appendix I).

Flight L/D data did disagree with the Langley 7- x 10-foot wind tunnel L/D curves in several cases. In the $\delta U_B = -21$ degrees, $\delta R_B = -10$ degrees configuration at 0.5 and 0.6 Mach numbers (figures 23 and 24, appendix I), flight data appear to reach maximum L/D at a much lower C_L than the wind tunnel and drop well below wind tunnel L/D's at the higher C_L 's. This data was obtained across the boundary of tip fin flow separation which is discussed in detail in the Tip Fin Flow Separation section.

Flight L/D data also differs from the Langley 7- \times 10-foot wind tunnel data in the approach configurations (figures 27 and 28, appendix I). Flight data is generally higher in L/D than the tunnel curve which is at the low boundary of the scatter. The full scale wind tunnel curve forms an upper boundary of the scatter so that the two wind tunnel curves form an envelope enclosing the flight data.

Cornell wind tunnel L/D points at the $\delta U_B/\delta e_L=-40/25$ degrees configuration, which were selected at flight trim angles of attack, were generally in good agreement with flight test data (figures 1 to 6, appendix I). The Cornell trim L/D curve at the $\delta U_B=-13$ degrees, $\delta R_B=-10$ decrees configuration was also in reasonably good agreement with flight data (figure 27, appendix I).

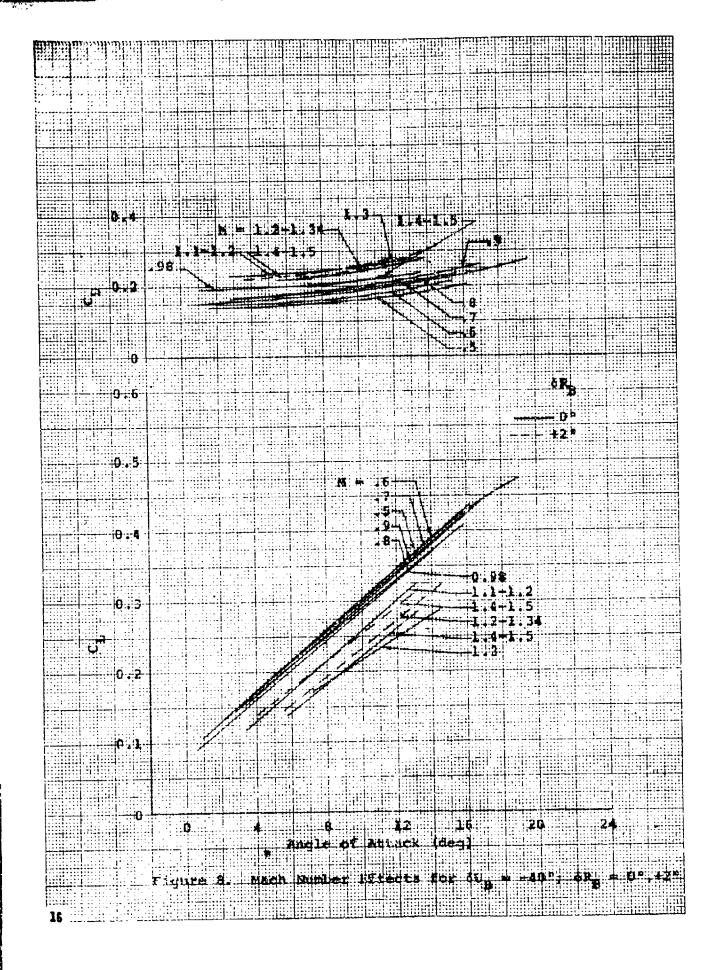
Flight gear down L/D data are compared to full scale wind tunnel gear down curves in figures 29 and 30, appoindix I. The Flight data generally exhibited higher L/D's than the wind tunnel curves. Flight data obtained with the canted gear door were generally slightly higher in L/D than those obtained with the perpendicular gear door.

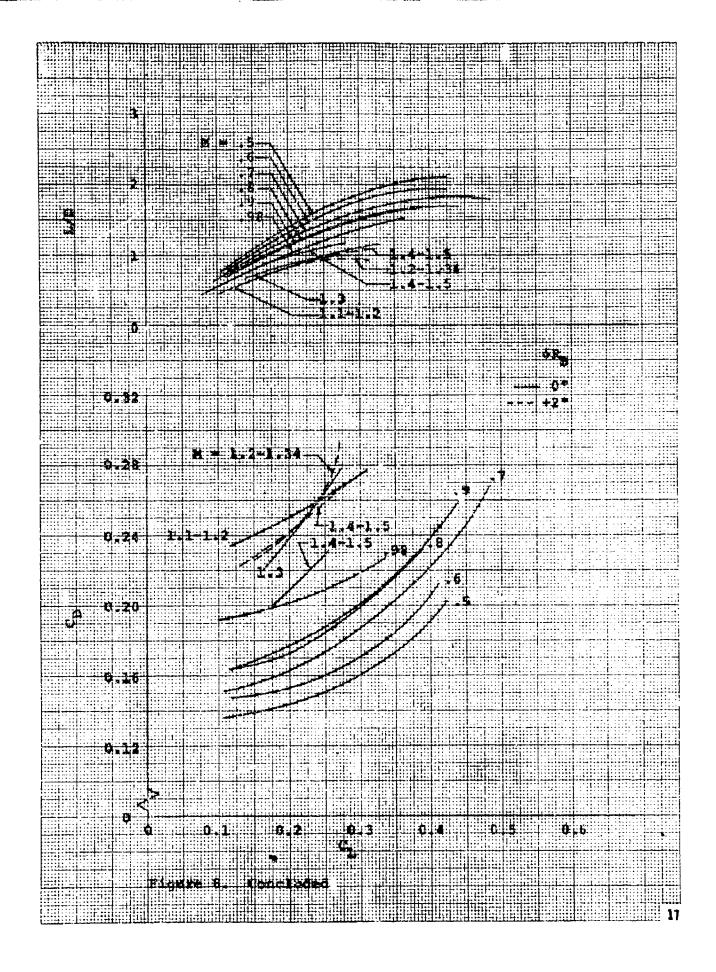
FAIRED FLIGHT TEST COMPARISONS

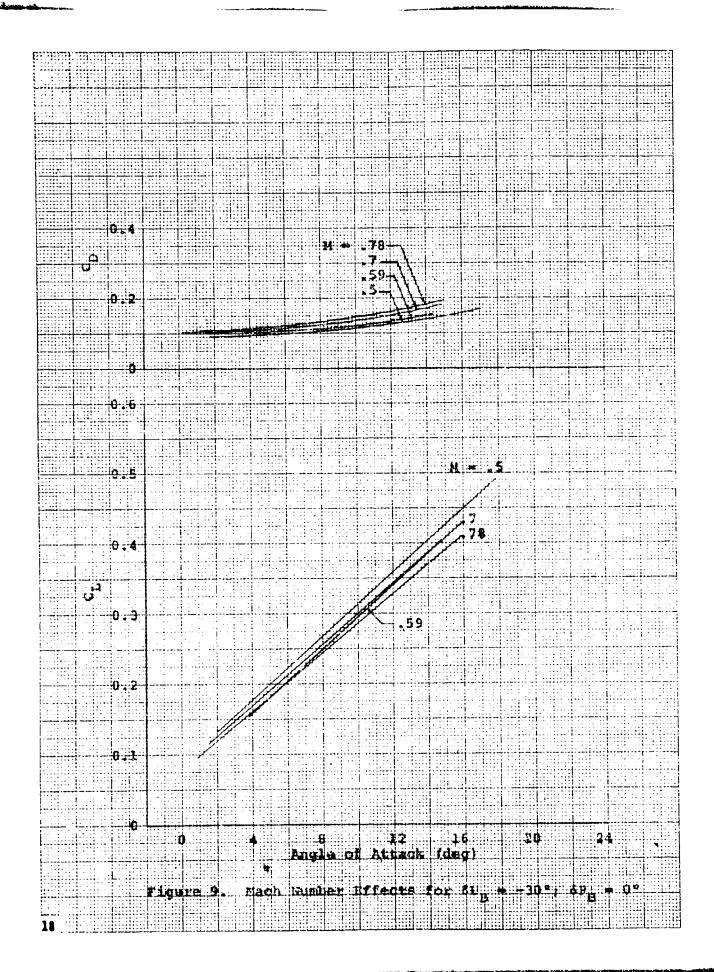
Paired flight data are presented in terms of C_L and C_D versus ε , and L/D and C_D versus C_L . Faired flight values of C_L^2 versus C_D are also presented. Comparisons are made to illustrate Mach number, wedge angle, and rudder bias effects on performance characteristics. In addition, the effects of tip fin flow separation and landing gear deployment are analyzed.

MACH NUMBER EFFECTS

For two transonic control surface configurations ($\cdot U_B = -40$ degrees and -30 degrees with $\delta R_B = 0$ degrees) faired flight curves are compared for the entire Mach number range in which data was obtained, in order to show Mach number effects (figures 8 and 9).







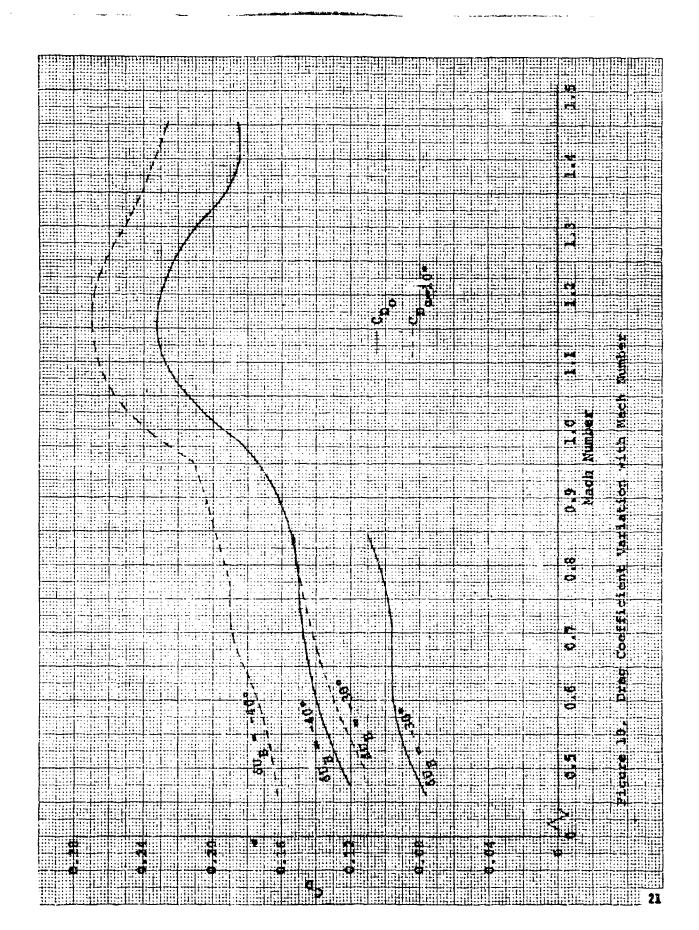
Lilt Coefficient

Data were obtained with -40 degrees upper flap bias for the entire Mach range of the program from M = 0.5 to 1.5 (figure 8). In the subsonic region from M = 0.5 to 0.9, there was little change in the CL versus curve as a function of Mach number. The actual data points fell within the same scatter band. The CL values for a particular trim angle of attack began to decrease slightly as transonic speeds were reached, and decreased greatly at Mach 1.0. Values of CL reached a minimum at about M = 1.3 and increased slightly at M = 1.4 to 1.5. The CL slopes did not change significantly with Mach number.

The same general trends are apparent but not as well defined in the $C_{\rm L}$ versus a data for the -30 degrees upper flap bias configuration (figure 9).

Drag Coefficient

Faired drag coefficient versus angle of attack curves and drag polars are presented in figures 8 and 9. Figure 10 summarizes the zero lift drag $(c_{\mathrm{D},c})$ variation with Mach number for the two transonic vehicle configurations (${}^{5}\text{U}_{\mathrm{B}}$ = -40 degrees and -30 degrees with ${}^{4}\text{P}_{\mathrm{B}}$ = 0 degrees). Zero lift (${}^{4}\text{S}_{\mathrm{B}}$) were obtained by extrapolating the nearly linear ${}^{6}\text{C}_{\mathrm{L}}$ versus ${}^{6}\text{C}_{\mathrm{D}}$ curves to $C_{\rm L}$ = 0. In addition, curves representing $C_{\rm D}$ at 10 degrees angle of attack $(CD_{i=10})$ are presented to illustrate the effect of drag due to lift throughout the Mach range and to show how total drag coefficient varied for a realistic flight angle of attack. Subsonically, zero lift drag increased slightly with increased Mach number. A drag divergence appeared at a Mach number of about 0.9. Maximum drag occurred at 1.15 Mach number and then slowly decreased with increasing Mach number. A comparison of the two drag curves with ${
m ^{17}U_{
m B}}$ = -40 degrees shows that drag due to lift for this configuration was practically constant throughout the Mach range. Values of $\mathrm{Cp}_{\mathrm{c=10}}$ were greater than values of Cp_0 by a constant ${}^{\circ}$ Co of about 0.04. For the other transonic configuration (Fig = -30) degrees; $\exists R_{\rm B}$ = 0 degrees), drag due to lift appeared to increase with Mach number subsonically. The supersonic values of ${\rm Cp}_0$ and ${\rm Cp}_{\rm c=10}$ were based on a very limited amount of data and were probably less accurate than sucsonic values. However, there was enough information to indicate the expected relationship between drag and Mach number.



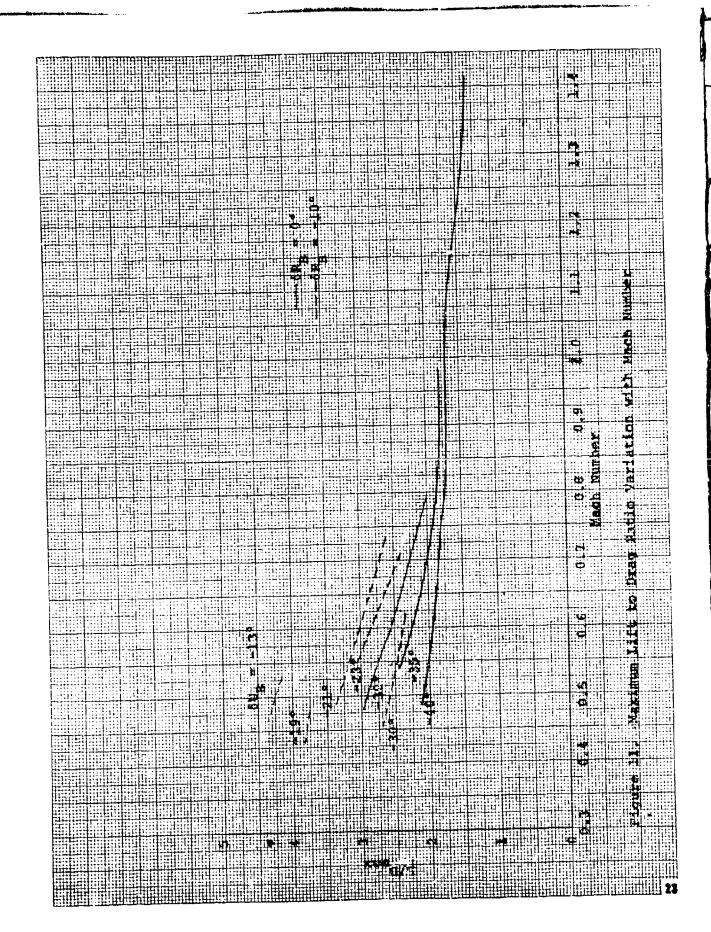
Lift-to-Drag Ratio

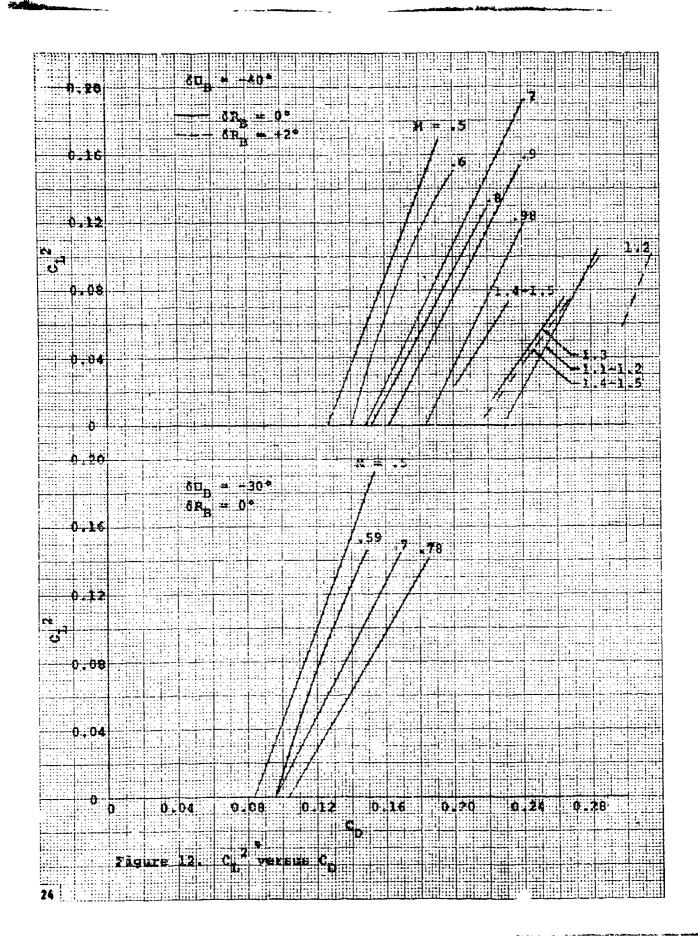
Faired L/D curves are plotted against C_L in figures 8 and 9. The L/D decreased with increased Mach number for both configurations. In figure 11, maximum L/D is plotted against Mach number for all fixed upper flap bias configurations for which performance data were obtained. Values of maximum L/D were obtained directly from the flight data in appendix I for those cases in which the flight maneuvers were taken to L/D_{max} . For those maneuvers in which L/D_{max} was not attained, flight lift and drag data were extrapolated to higher angles of attack to obtain the most plausible values of L/D_{max} . The curves show that maximum L/D decreased sharply with increased Mach number in the low subsonic region and leveled off in the transonic and supersonic regions. Figure 11 also indicates that this subsonic decrease in L/D_{max} was generally smaller at the higher upper flap bias configurations. The degradation in L/D_{max} with increased Mach number at low subsonic speeds was attributed to tip fin flow separation and is discussed in the section of the report with that title.

CL2 Versus CD

Faired plots of C_L^2 versus C_D are presented in figure 12 for the $\delta U_B = -40$ and -30 degrees configurations. In the $\delta U_B = -40$ degrees configuration, the slopes of the resulting straight lines $(\frac{d}{dC_L})^2$ are 2.6 in the subsonic region (M = 0.5, 0.6), decreasing to 1.9 in the transonic region (M = 0.70 the 0.98) and ranging from 1.89 to 1.40 at supersonic Mach numbers and zero 1 der bias. Supersonic data at +2 degrees rudder bias were so limited th. faired $\frac{d(C_L^2)}{dC_D}$ slopes were not of sufficient accuracy to show trends. At the smaller upper flap setting ($\delta U_B = -30$ degrees), the C_L^2 versus C_D slopes had a value of about 2.7 at M = 0.5 and 0.59 and showed a steady decrea 2 with increased Mach number.



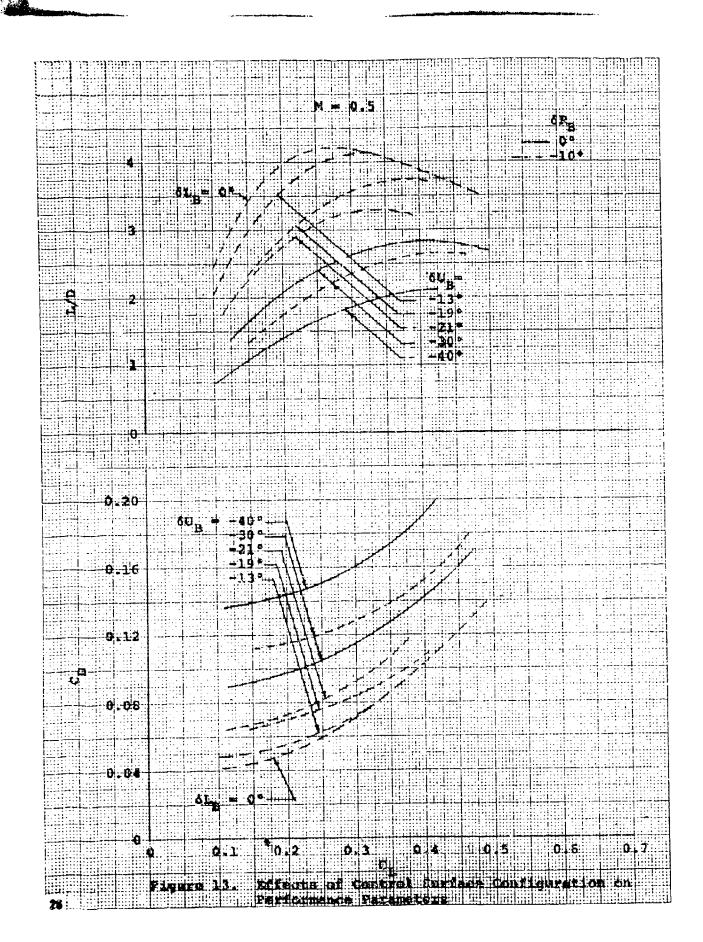


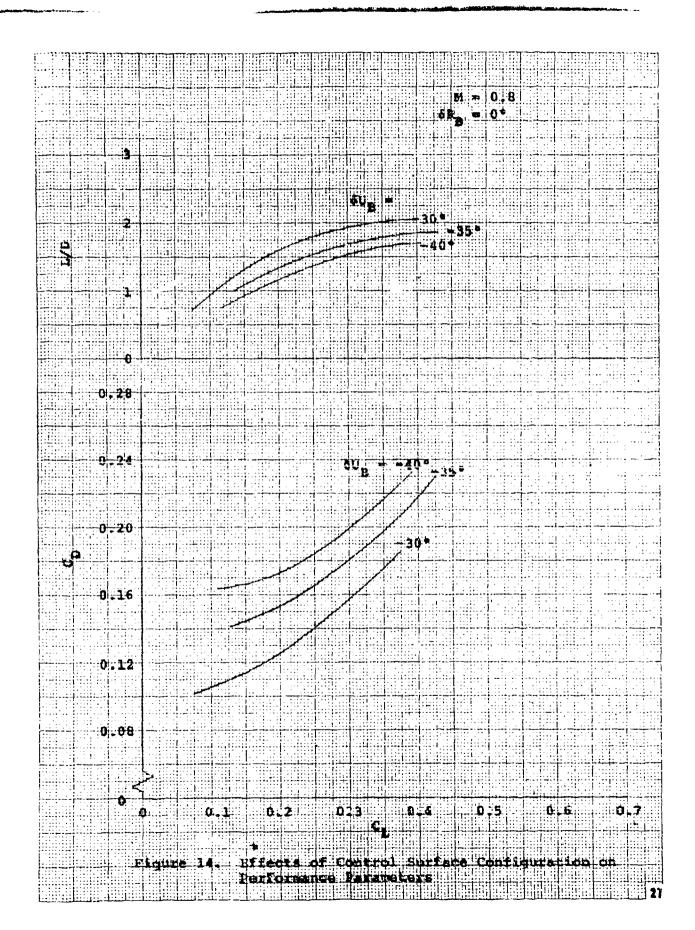


WEDGE ANGLE and RUDDER BIAS EFFECTS

The variable features of the X-24A flight control system allowed performance data to be collected over a reasonably wide range of upper flap, lower flap and rudder bias positions. Representative faired data at 0.5 and 0.8 Mach number are shown in figures 13 and 14. At 0.5 Mach number, trim test data were obtained at upper flap bias settings of -40 and -30 degrees with 0 degrees rudder bias; and at upper flap bias settings of -30, -21, -19 and -13 with -10 degrees rudder bias. Data were also obtained while controlling on the upper flap ($\delta L_{\rm B}=0$) at -10 degrees rudder bias. Because of the predicted poor stability levels at high Mach numbers and small upper flap settings, performance data were limited to -30, -35 and -40 degrees upper flap bias at 0.8 Mach number. These variable control settings were extremely effective in altering the performance of the X-24A especially at 0.5 Mach number where maximum L/D was doubled by closing the upper flaps from -40 to -13 degrees.

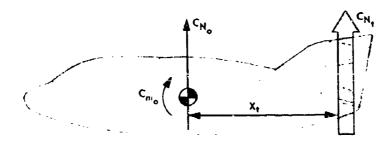
The primary effect of the flap bias feature was to alter the base area and thus, the base drag of the vehicle. When the upper flap was extended a pitch-up moment was produced which was counteracted (either by the pilot or the control system) by extending the lower flap the proper amount to remain in trimmed flight. When the upper flap was retracted the lower flap was also retracted to remain in trim. The change in base area associated with upper and lower flap changes was therefore additive and related to the change in the total angle between the upper and lower flaps, or wedge angle (ow). Movement of the rudder bias surfaces on the tip fins did not directly alter the base area or base drag of the vehicle. It did result in a strong indirect effect since the rudder position influenced the pressure in the area of the upper flaps. When the rudders were biased inboard a large pitchup moment was produced (similar to extending the upper flap). This moment was counteracted by either opening the lower flap or by closing the upper flap, thereby altering the base area and thus the base drag.





Wedge Angle Effects on Trim Normal Force Coefficient

Since the centers of pressure of the upper flap, lower flap and radders were essentially the same distance from the center of gravity it follows that for trimmed flight, the trimmed normal force coefficient (C_N) would be independent of the flap or rudder bias configuration (figure 15). This effect was apparent in the analysis of all wind tunnel data and is confirmed by the flight data shown in figures 16 through 18. Within the scatter of the test data the trimmed C_N curve is unaffected by changes in the upper flap bias setting or rudder bias setting. The lines represent the data from the Langley 7- x 10-foot wind tunnel for all flap configurations. The slopes of C_N versus angle of attack were less for flight test data than for wind tunnel data.



 $C_{m_0} = X_1 - C_{N_1} = 0$ FOR TRIMMED FLIGHT $C_{N_{\mbox{\footnotesize{TRIM}}}} - C_{N_0} + C_{N_1}$

WHERE Cm Cm FOR ZERO CONTROL SURFACE DEFLECTIONS

 $C_{N_{\alpha}} = C_{N}$ for zero control surface deflections

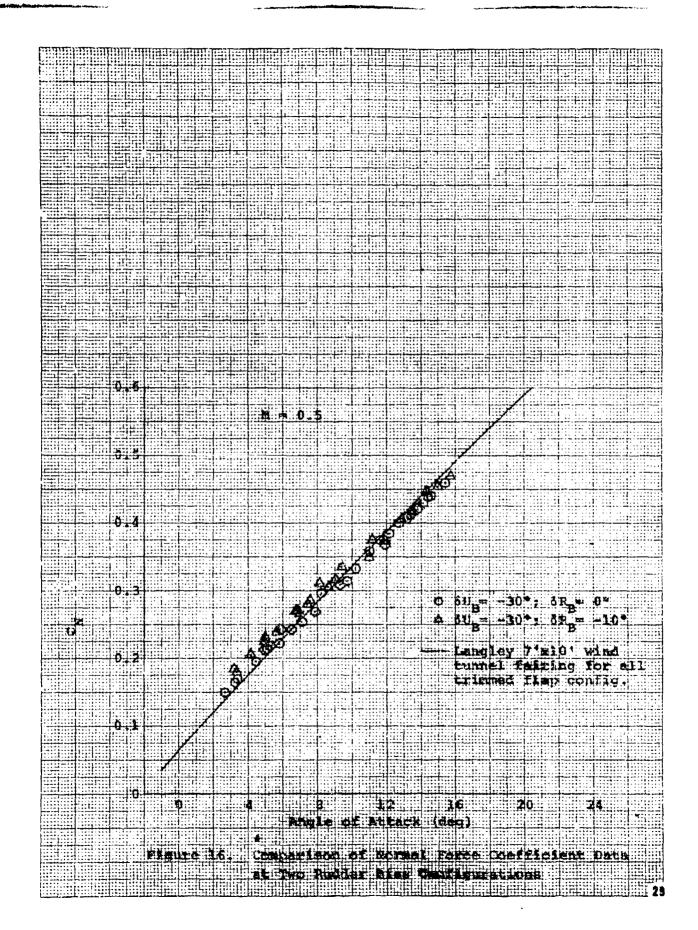
X_t = TAIL LENGTH (approximately the same for upper flap, lower flap and rudder bias)

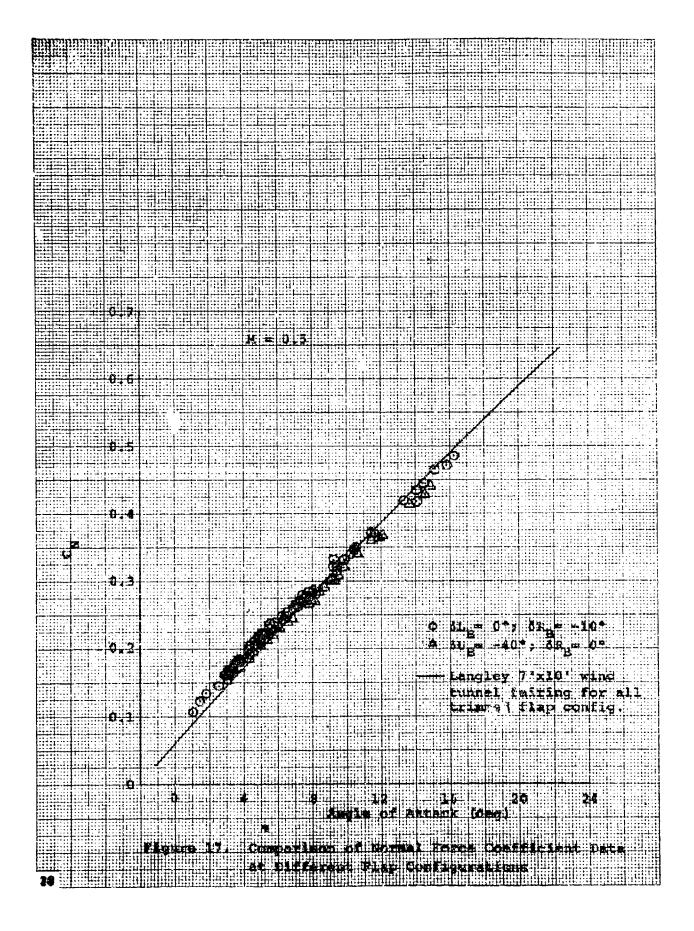
CN. - CN AT THE TAIL REQUIRED FOR TRIM

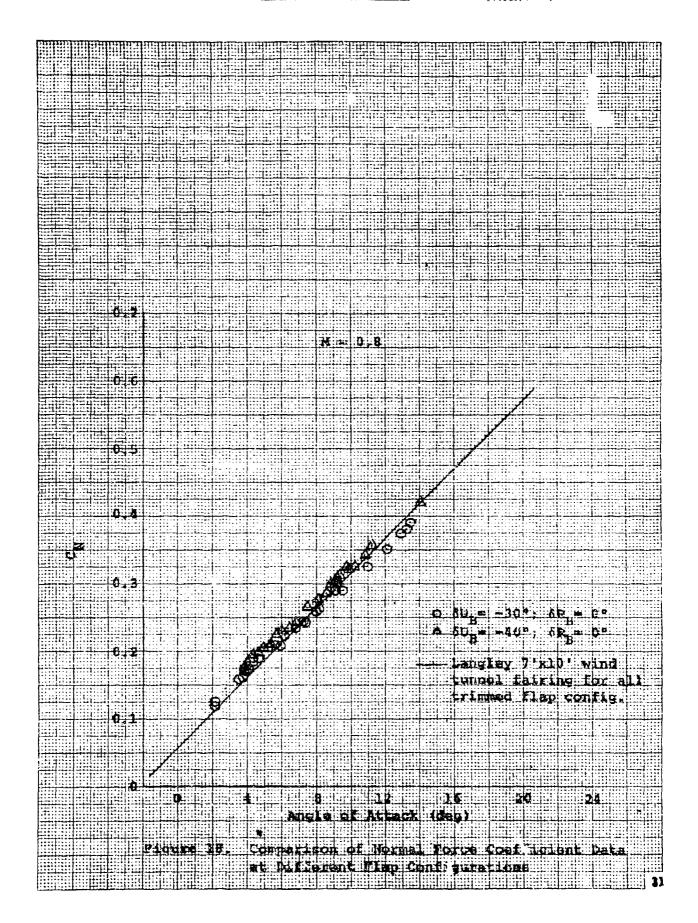
NOTE:

The magnitude of $C_{N_{\uparrow}}$ required to produce trimmed flight is the same regardless of which control surface is used.

Figure 15 Induced Normal Forces for Trimmed Flight



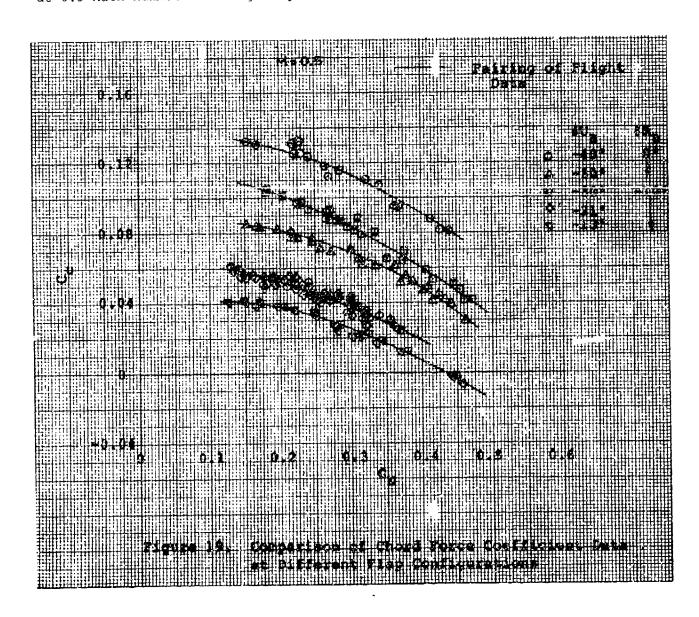




Wedge Angle Effects on Chord Force Coeilicient

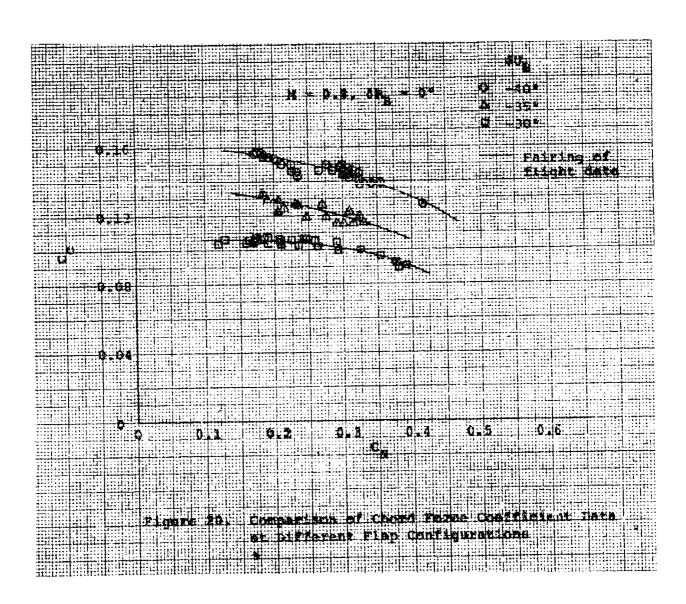
Early analysis of wind tunnel data indicated that chord force coefficient (C_C) for a particular Mach number and trimmed C_N could be related directly to total wedge angle, regardless of the positions of the individual upper or lower flaps. Furthermore, this relationship appeared to be parabolic [$C_C = f(\delta w^2)$].

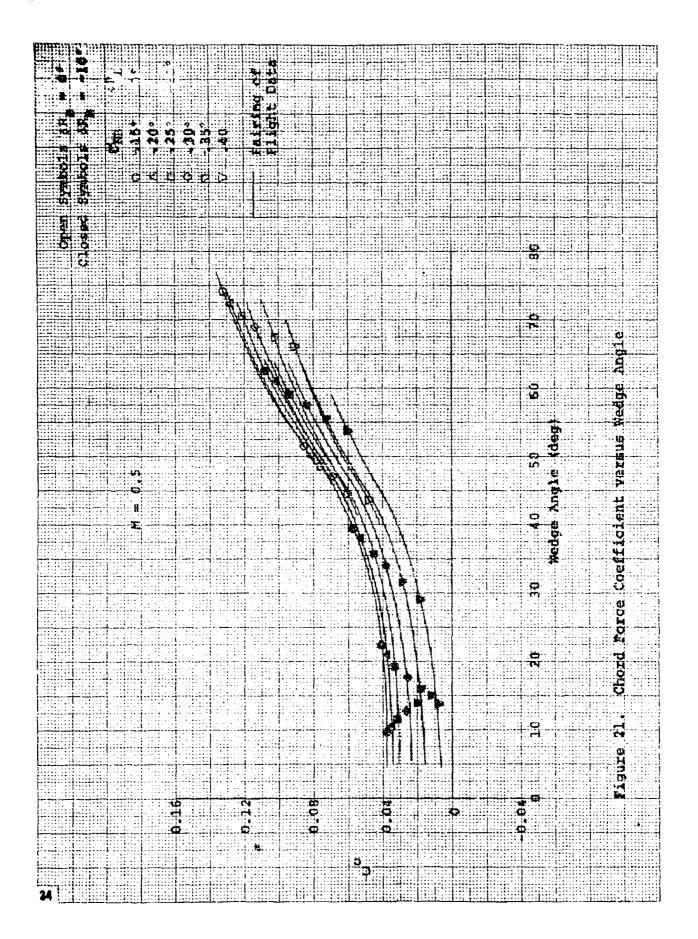
The faired lines of figures 19 and 20 at 0.5 and 0.8 Mach number are cross plotted as $C_{\rm C}$ vs 6w for constant trimmed $C_{\rm N}$ in figures 21 and 22. These curves are essentially parallel and do exhibit a parabolic shape at 0.5 Mach number and wedge angles below 50 degrees.

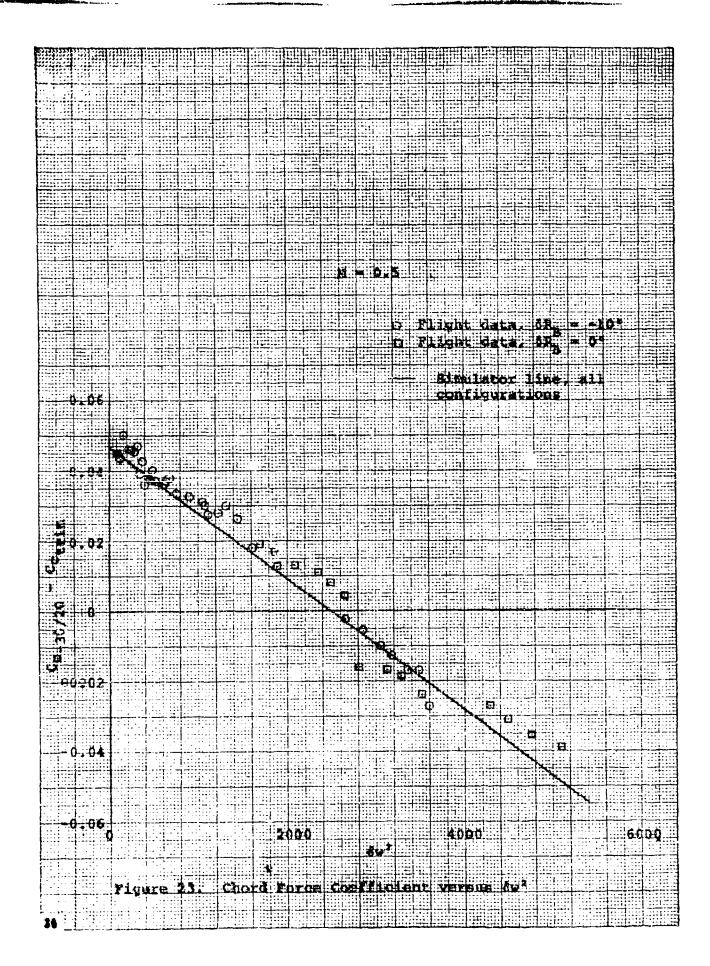


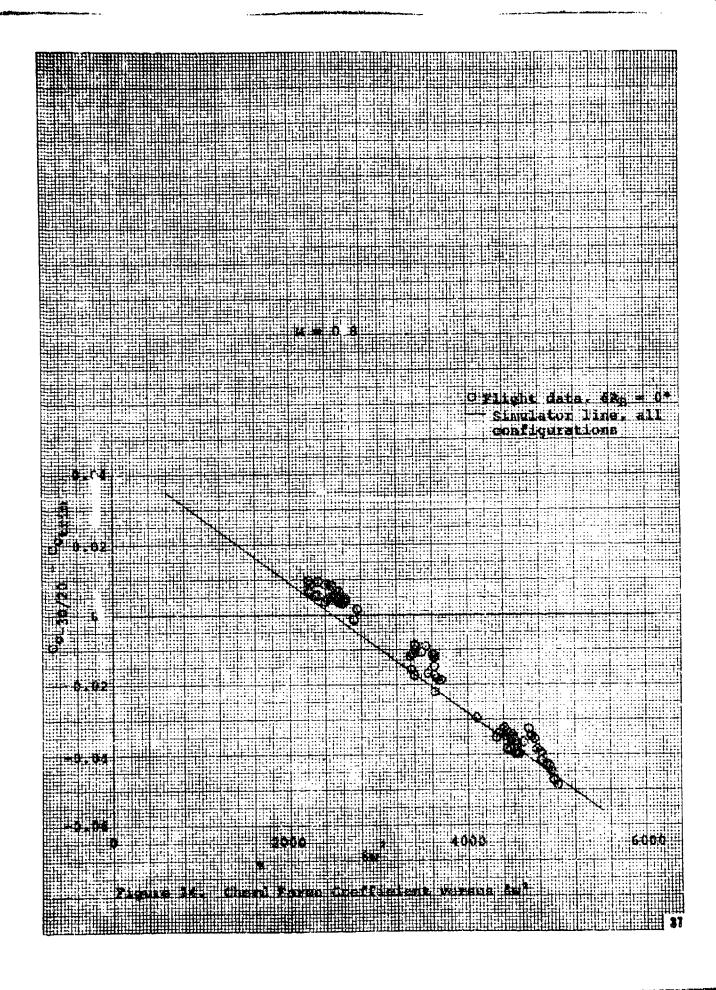
The flight data points were normalized with respect to a using wind tunnel values of $C_{\rm C}$ for the -30 degrees $\delta U_{\rm B}$, +20 degrees $\delta e_{\rm L}$ configuration. These points are shown plotted versus δw^2 in figures 23 and 24 and confirm the parabolic variation of $C_{\rm C}$ with δw . The straight line shown on each of these figures is the slope established from prior analysis of wind tunnel data and was used in the X-24A simulator.

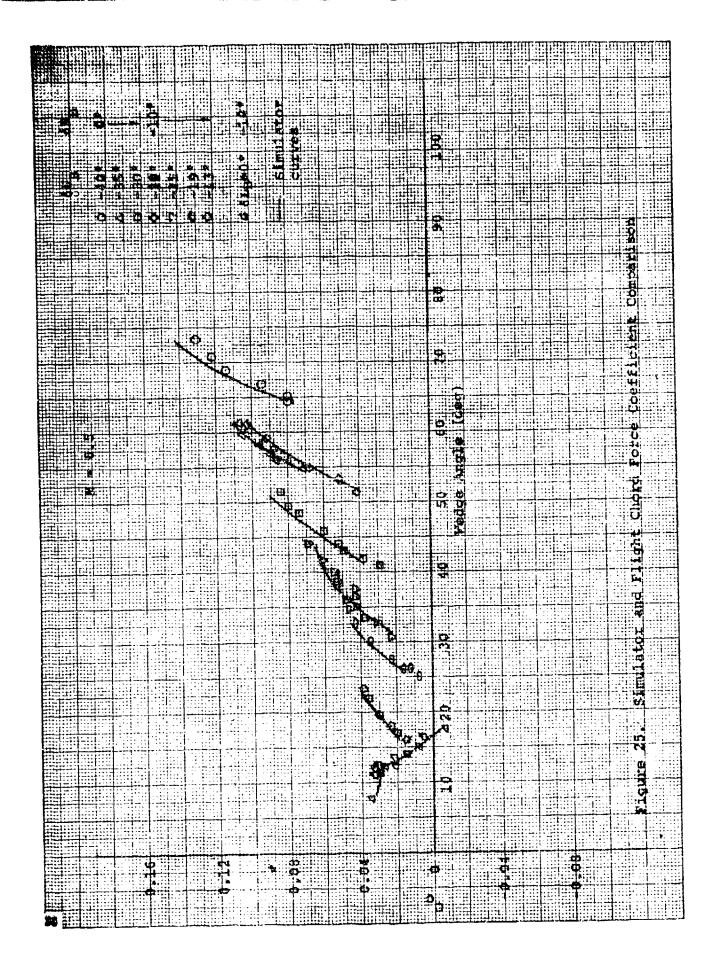
Flight data points of trimmed $C_{\rm C}$ at 0.5 Mach number are compared with simulator values of total trimmed $C_{\rm C}$ for the same flight conditions in figure 25. The simulator values compare well with flight data except at the high wedge angles where simulator values were somewhat higher than the flight data.











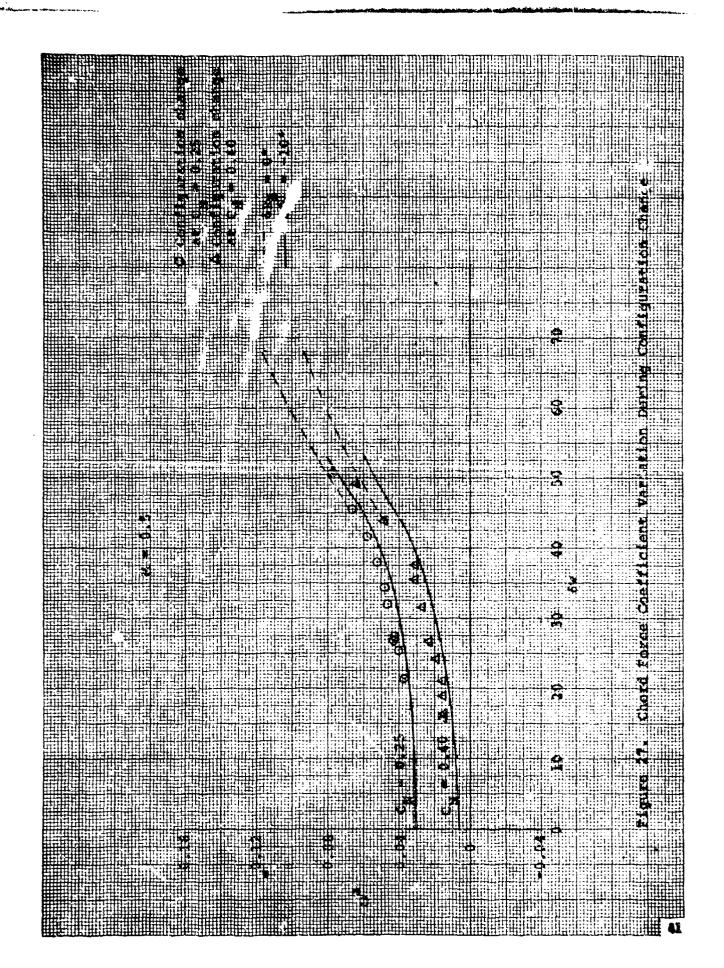
Wedge Angle and Rudder Blas Effects on Liff-to-Drag Ratio

The combined effects of Sw and SRB on the lift-to-drag ratio at 0.5 Mach number are shown in figure 26. The indirect effect of rudder bias on the performance characteristics is evident by comparing the zero degree and -10 degrees &RB lines in this figure. The vertical separation of these lines indicates a small apparent improvement in performance (approximately 0.1 L/D) associated with biasing the rudders inboard at constant wedge angle (direct effect of the rudders). A comparison of the two -30 degrees &UB points, however, shows that this small positive increment is more than counteracted by the increase in wedge angle associated with extension of the lower flap to return to trim. The net result is a decrease in L/D of approximately 0.3. Note that the vehicle could also be retrimmed by retracting the upper flap and thereby decreasing the wedge angle and increasing the L/D. This is indicated by the solid point in figure 26 which is an interpolated point showing L/D at 0.3 CN resulting from a configuration change from 0 to -10 degrees SRB, while holding the lower flaps fixed at 17 degrees and trimming with the upper flaps (decreasing from -30 to -22 degrees).

Speed Brakes

Prior to flight nine, the automatic rudder bias feature was reprogrammed to follow the upper flap bias so that the rudder bias and upper and lower flap bias trim changes would tend to cancel (reference 1). Thus the flaps and rudders were electrically linked together and could be extended or retracted with a minimum trim change apparent to the pilot. This "speed brake" feature was used by the pilots thereafter during the landing approach. Flight data points obtained during the extend or retract cycle for two values of trimmed $C_{\rm N}$ (two α 's) are shown in figure 27 and compared with previous data obtained from pushover-pullups at constant rudder bias settings. The total variation in L/D available through the use of the upper flap bias "speed brake" is shown versus $C_{\rm L}$ S/W and $V_{\rm e}$ in figure 28. The shaded portion of figure 28 presents the actual amount of speed brake used in the program.

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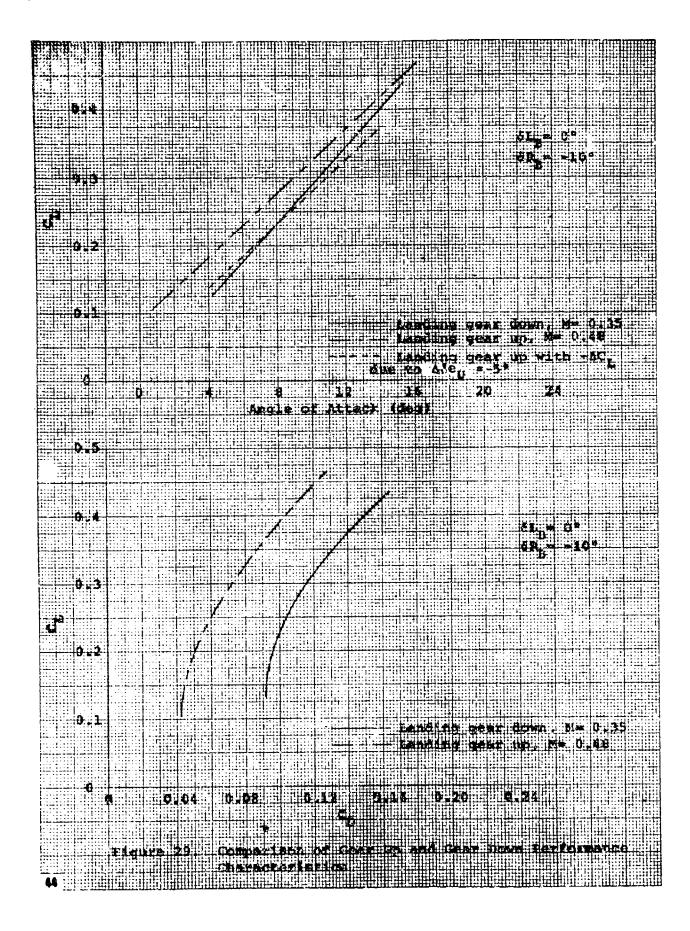
LANDING GEAR EFFECTS

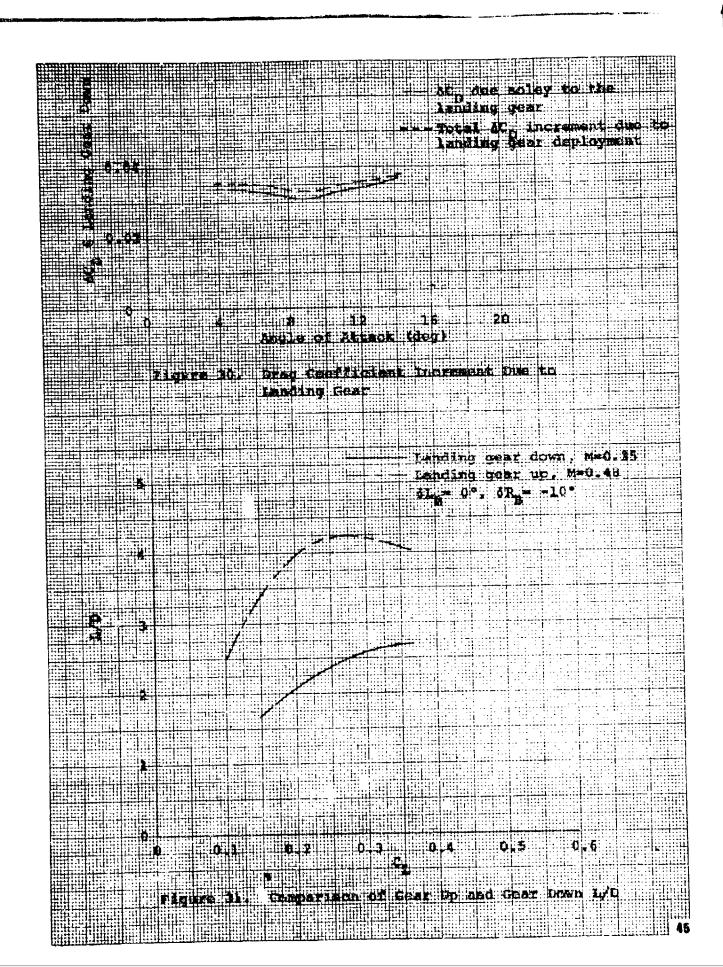
Figures 29 and 31 compare gear up and gear down faired performance test data in the $\delta L_B=0$ degrees, $\delta R_B=-10$ degrees control surface configuration. Gear down curves exhibit a greater lift curve slope, increased total drag and significantly lower L/D's when compared to corresponding gear up curves. The differences that were apparent between the gear up and gear down data were not due solely to the landing gear. Since deployment of the landing gear produced a nosedown trin change, an increase of approximately five degrees of upper flap was required to remain at the same trim lift coefficient.

The gear down lift coefficient versus a curve (figure 29) is below the gear up curve and has a steeper slope by 21 percent. The reduced gear down lift coefficient at the same angle of attack was a result of the down load at the tail (increased upper flap setting) required to compensate for the trim change. (This increment as applied to the gear up curve is shown as the dashed line in figure 29 and is based on flight measured values of $C_{N, \epsilon}$, reference 5.) The apparent increase in lift curve slope is probably due to ground effect since gear down data were obtained during actual landings in which the gear was extended between 40 and 120 feet above the ground. The data at the higher angles of attack and slower speeds were obtained closer to the ground and were therefore more strongly influenced by ground effect.

The total drag increment observed was the sum of the landing gear drag and the increased base drag resulting from the longitudinal trim change. To obtain the drag increment due to the landing gear alone the difference between the gear up and year down curves in figure 29 was obtained and a correction applied for the five-degree difference in wedge angle using data in figure 21. (Notice that this correction was quite small in the wedge angle range of 15 to 20 degrees associated with landings.) The resulting drag coefficient increment due to landing gear alone was between 0.030 and 0.035 (figure 30).

Gear up and gear down L/D data are compared in figure 31. The deployment of the landing gear caused a 37 percent reduction in the maximum ${\rm L}/{\rm D}$.





TIP FIN FLOW SEPARATION EFFECTS

Tup fin flow separation was noticed on early X-24A flights. A study was made to determine at what conditions separation occurred. The results of this study are reproduced in full in appendix V.

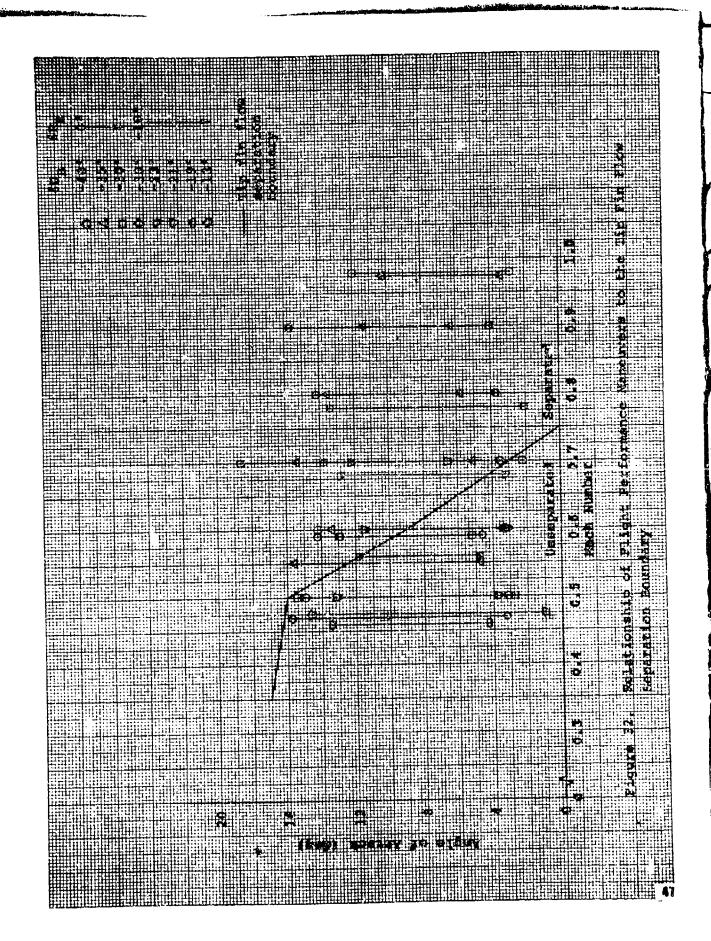
As a result of this study, a tip fin flow separation boundary was generated and presented in terms of apparent angle of attack for local tip fin stall versus Mach number. Analysis of wind tunnel data indicated that as upper flap setting was increased, static stability also increased thus apparently reducing the severity of the separation effects. However, the angle of attack at which separation occurred remained the same for all upper flap settings.

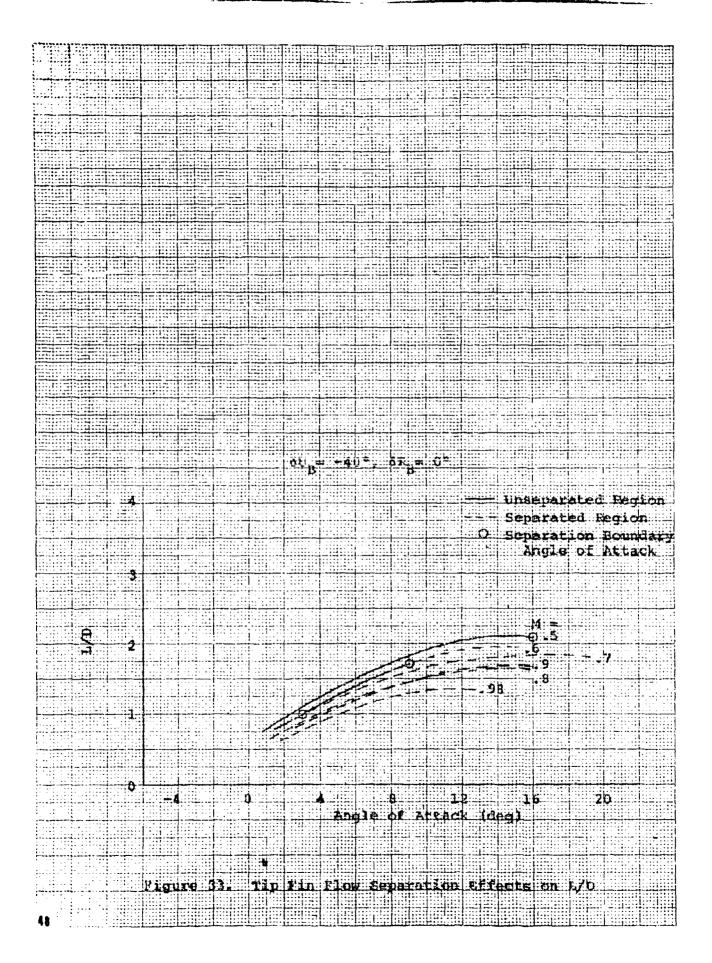
Figure 32 presents a comparison of this boundary with the full range of flight performance subsonic and transcnic data obtained during the program. If the boundary is correct, a comparison of data in the unseparated and the separated regions, should show decreased performance in the latter. Also, the effects should become greater as upper flap setting is decreased. Performance flight data appear to substantiate this.

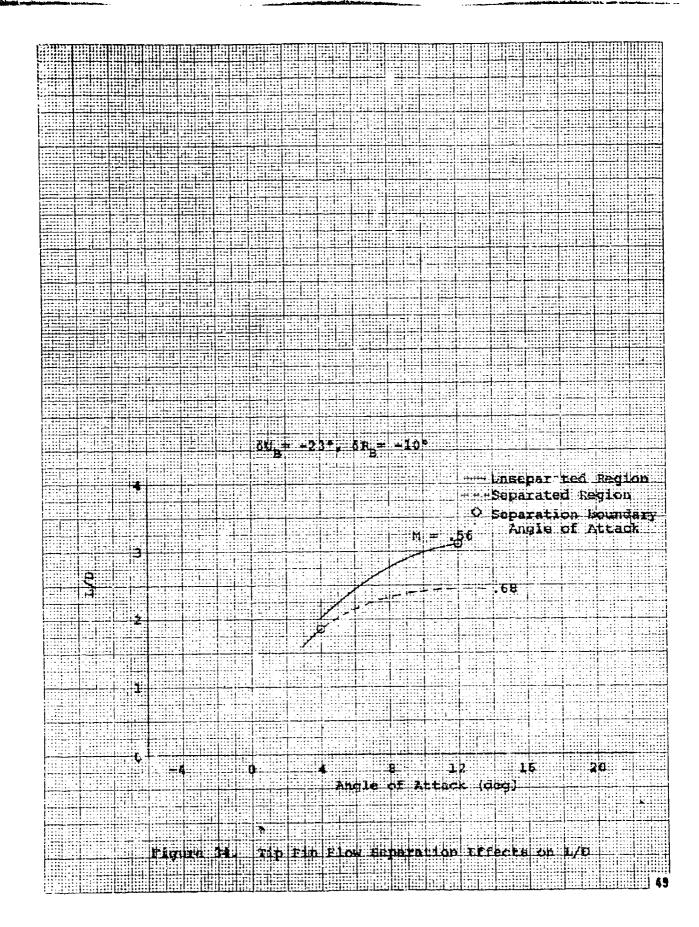
Figures 33 through 35 show faired flight data in terms of L/D versus angle of attack for three upper flap settings for a range of Mach numbers. The solid part of each curve represents data obtained in the unseparated region. The dashed part of each curve represents data obtained at angles of attack above the separation boundary (separated flow on the tip fins). With the upper flaps set at -40 degrees (figure 33), there was a gradual decrease in L/D_{max} with increased Mach number, probably due mostly to Mach number effects. Data of tained with the vehicle in the other two transonic configurations (${}^{\circ}U_{B}$ = -35 and -30 degrees) exhibited similar trends and were not shown. However, with the upper flaps set at -23 and -21 degrees, drastic differences, both in maximum L/D and in the total shape of the L/D curve, are apparent between data obtained in the separated and unseparated regions (figures 34 and 35). The maximum L/D is drastically reduced and a significant flattening of the curve occurs in the separated region. The dramatic differences between the 0.5 and 0.7 Mach number curves at $iU_{\rm B}$ =-21 degrees, and the M = 0.56 and 0.68 curves with $^{\circ}$ U_B = -23 degrees, are very similar. At 0.6 Mach number with a -21 degrees upper flap setting, the separation boundary was crossed about midway through the maneuver. The data show a definite break at the boundary angle of attack and approaches the totally separated curve at 0.7 Mach number. Therefore, the flight performance data appear to correlate very well with the separation boundary.

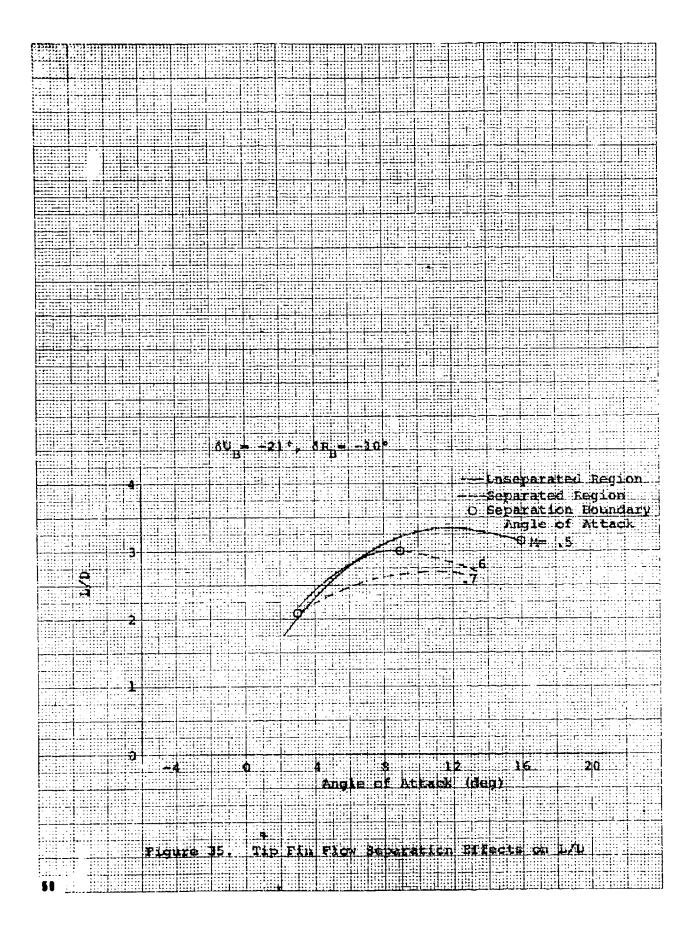
Figure 36 presents X-24% flight data at three upper flap settings, along with small scale and full scale wind tunnel data in terms of maximum L/D versus Mach number in the subsonic to transonic region. The data compare very well except for low speed data in the $\delta U_{\rm B} = -21$ degrees upper flap configuration. Wind tunnel performance results were slightly optimistic with respect to Mach number or angle of attack in predicting L/D in the transition region between separated and unseparated flow. This trend was also found with the BL-10 lifting body (reference 9).

Due to the significant effects of this type of separation, it is important that wind tunnel studies accomplished prior to the test flight program be analyzed for these trends. Wind tunnel stability measurements were much more significant and gave a much clearer indication of tip fin flow separation than did wind tunnel performance measurements (appendix V).









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CONCLUSIONS

A large amount of performance data was obtained during the X-24A flight test program which confirmed the design goals of a subsonic L/D_{max} of over 4.0 and a supersonic L/D_{max} of approximately 1.25. The performance characteristics of the vehicle were defined for a variety of vehicle control surface configurations over a Mach number range of N = 0.26 to 1.50 and at angles of attack from 1.5 degrees to 19.6 degrees.

Different wind tunnel tests showed discrepancies with each other in their predictions. Generally, however, flight test data exhibited lower lift at the same angle of attack, and lift curve slopes that were less than most wind tunnel predictions. Flight test drag coefficient data were also generally slightly less than wind tunnel data with the result that flight L/D's were similar to wind tunnel predictions.

Some Mach number effects were apparent in the flight data. Flight lift curve slopes did not change appreciably with changing Mach number. However, lift coefficients at the same angle of attack showed significant decreases with increased Mach number above M=0.9. Zero lift drag coefficient data were derived and showed the expected transonic drag rise. The reduction in L/D with increasing Mach number which occurred in the low subsonic Mach range $(0.5 \le \text{M} \le 0.7)$ was attributed to tip fin flow separation. Plight performance data substantiated the tip fin flow separation boundary which was developed from hinge moment data and tuft photos early in the flight test program.

The effects on performance of control flap configuration were found to be a total wedge angle effect. This was manifested by a change in base area which contributed a corresponding change in base drag. Flight chord force coefficient data at low subsonic Mach numbers were found to be a parabolic function of wedge angle which confirmed wind tunnel predictions.

The drag coefficient increment due to the landing gear alone ranged from 0.030 to 0.035. The total effect of landing gear deployment (landing gear plus an increase in wedge angle due to trim change) produced a reduction in $\rm L/D_{\rm max}$ of 37 percent in the final gear down landing configuration ($\rm ^1L_{\rm R}$ = 0 degrees).

The speed brake feature of the X-24A control system, which was developed during the test program, provided the capability for a variation of $L/D_{\rm max}$ from 4.25 to 2.10 at Mach numbers below 0.8.

APPENDIX I FLIGHT TEST AND WIND TUNNEL TRIM PERFORMANCE DATA

Table 1

100 OF FLIGHT TEST PERFORMANCE MANEUVERS

Upper	1			Angle of				- 1		1	
Flar	Ruider		Lower Flap	Attack Fange	flight	- 6	Gross Weight		Ÿ	ž į	ž (
Bias (deg)	(deg)	Mach Nurb r Range	Range (deg)	ਵਗਰਾਜ਼ (ਹੋਵਰੂ)	Number	R_x10 ⁻⁶	(15)		(in.)	(in.)	(pet)
				Transonic (onflaurai	LIGUS					
-40	0	.466 to .501	24, to 33.5	3.6 to 14.9	23	38.1	6,553	155.40	512	27.:8	59.31
	+4.20	.509 tc .534	24.6 to 30.4	3.8 to 11.5	2.7	34.5	6,157	156.16	. 141	27,39	14.57
	0	572 to .605	26.1 to 33.1	3.5 : : 14.2	2.3	33.2	6,558	155.40	612	27,19	56.31
i '		.601 to .754	24.5 to 33.5	1.5 to 15.1	16	34.4	7,404	150.90	326	27.71	54.63
1		.704 to .746	23.6 to 27.9	8.5 to 18.8	20	38.5	7,301	152.60	277	27 74	55.36
i '	+2.20	.695 to .733	24.8 to 30.2	3.8 to 13.5	2.7	25.⊁	6,157	156.10	.141	27,66	56.57
	0	.763 to .851	27.8 to 31.1	3.8 to 14.2	20	35.3	7,301	152.80	277	27.74	55.36
1		.847 to .877	24.6 to 30.4	4.1 to 13.8	18	25.6	7,220	151.40	SR3	27.20	56.79
		.866 to .941	24.6 to 29.8	4.2 to 13.4	20	30.5	7,301	152.80	277	27.74	55, 36
1		.941 to 1.308	25.9 to 32.2	2.8 to 10.6	19	19.3	7,289	152.40	531	27,55	55.20
ļ		.955 to 1.0!0	23.4 to 30.4	3.6 to 12.0	20	20.1	7,301	152.80	-,277	27.74	55.36
1	ļ	1.240 to 1.334	36.8 to 21.0	7.8 to 17.4	20	15.6	7,301	152.80	277	27.74	55.36
	ĺ	056 to 1.261	19.9 to 29.3	4.4 to 11.2	2.5	14.9	0,538	155.40	(12		35.31
j		1.414 to .486	17.1 to 20.4	8.4 to 10.8	23	17.5	6.558	155.40	.612	27.;A	55.31
1	+1.25	1.341 to 1.452	19.4 to 25.0	4.9 to 10.3	25	16.6	6,445	156.10	.184	27.0	56.55
— —	+2.00	1.671 to 1.337	19.9 to 23.6	6.5 to 13.4	26	12.5	5,324	157.80	121	27.03	57.15
-35	0	.518 to .597	20.2 to 25.4	6.5 to 12.5	11	39.6	6,394	156.20	959	25.76	559
1	1	.523 to .539	18.5 to 27.2	4.7 to 15.6	14	29.9	6,430	116.00	143	26,93	56.54
1	ļ	.545 to .572	23.7 to 26.2	5.2 to 13 0	9	31.9	6,336	159.20	4:8	27.71	57.75
1		.571 to .644	20.7 to 27.5 29.0 to 22.5	11.3 to 15.1	9-	39.1	6.407	1.5.11	413	27.21	37.35
}	l	.689 to .757	21.1 to 25.7	5.1 to 15.4		32.5	6,194	156.23	- 19 1	21 7	35 58
ł		.719 to .741	22.2 50 25.4	5.i to i2.6		23.5	(,)94	150.2	533	49.75	54 , 54
1	İ	.734 to .756	21.8 to 26.;	5.3 to 11.2	/1	18.2	L,559	156.42	790	20,30	15.81
1		.754 to .864	23.0 to 26.0	6.3 to 11.3	12	25.9	6,463	155.70	.033	. 5. 97	56.42
		.755 to .855	21.7 to 24.9	7.0 to 13.7	14	27.0	5,430	156.00	.025	25,83	56.54
1	i	.859 to .955	23.7 to 26.0	6.5 to 11.5	14	23.2	6,410	156.00	.623	26.83	56.54
i	ľ	.662 to .987	21 3 10 26.8	3.3 to 10.2	14	17.3	1,120	14 4 . 110	174	27 82	54.33
- 30	1	.496 to .534	11.1 to 21.5	2.7 to 15.4	20	36.6	5.335	1:5.20	.21.	26.91	37.34
1	1	.587 to .591	14.1 to 20.2	5.2 to :2.9	7	32.5	6,83	158 59	+.15	2 . 56	51.45
		.666 to .684	16.4 to 21.1	4.0 to 12.9	8	34.5	6	158. 9	151	77,73	57,45
		.664 to .774	i5.3 to 22.0	2.3 to 11.9	19	42.1	7,289	152.40	54	3.5 55	15.2)
1		.750 to .805	16.6 to 22.1	4.1 to 13.5	19	33.7	7,289	150.46	551	2.35	1 5.29
				Sabsonio Co.	ntigurati	ogs.					
- 30	-10	.421 to .517	21.1 to 31.1	3,2 to 15.8	19	43.5	6,373	158.36	+1345	2	\$2.24
Ì	ŀ	.571 to .581	22.4 to 24.6	4.7 to 12.7	- 6	10.5	6,277	158.61	7.121	21.08	3.33
	1	.587 to .594	22.3 to 28.3	6.5 to 14.5	7	31.2	6,.80	158.59		27,68	57,45
-23]	.556 to .569	11.7 to 21.2	4.8 to 11.8	5	27.5	6,294	158.68	:37	27.72	57.55
-23.5	.]	.671 to .693	14.7 to 22.1	3.2 to .2.4	2	32.3	0.35	152.50	4.14.	27,62	50.20
-21	1	,483 to .342	9.9 to 19.9	2.6 to 13.1	2	31.9,45.5	այ, երձ	150.15	. 1±0	27.92	23
ł	İ	.574 to .614	11.8 to 19.0	7.1 to 13.4	<u> </u>	35.3	6,459	162.58	1.5	23.00	38.27
Ĺ	1	.660 to .718	11.5 to 14.6	6.5 to 17.2	1	33.1	0.350	6		200	Na. 36
<u></u>	<u> </u>	.443 to .474	6.8 (0.25.7	4.2 to 11.5	5	38.1	6,294	176.83	1.13	17.71	17.55
			S ,05	som a Approach	Contigui	at i sos					
-13	-19	.472 to .491	3.0 to 9.4	1.7 (0.14.6	3	36.5	6,353			2 .35	18.21
1	1	.150 to .478	1.6 to 8.4	1.9 to 5.5	13	30.1	1	V . 12		1 7 7 2	6.54
L]	.494 to .521	2.5 to 8.0	0.9 to 4.5	25	51,9,79.3		.5c.10	, -:	125,00	35.
-9.5		.477 to .496	11 ₁₈ = 0°	2.8 to 7.8	19	44,4,64.5	6,301	189.00	+ , , 46	10.22	57.24
-12.:				I	l	L	1		1	L	
-8.8	1	.450 to .517	11.g + 2.0	1.0 to 16.3	23	41.9	0.335	158.22	.21.	20.31	177,34
1:7.7	,]								Į	ì	
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 		1-301-35		. c Landing Gra			7	1.0.10	T	27.62	, , , ,
21	~10	.301 to .317	3.4 (12 7.1	8.6 10 12.7		17.5	6,359	100.58	1	27.54	58.20 .6.75
- 21	-	.256 to .323	0.4 (0.5.8	10.7 to i .4		45.5	6.431	161 31	143	-	
-15.c	·}	.250 to .35!	11 ₂₁ = 0	9.4 to 15.0	•	"'	6,290	128 -3	- 15.	"	
- 21.	<u>"</u>		ļ	ļ	↓	 	↓	 		 	∤
-15	s :	.2°C to .359	:1. _{is} = 0	5.3 to to 3	9	50.4	6,462	178.33	1 - 116		57,38
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	7	De to .415	L _B = 0	4. 1 to 10.7	1 3	54.2	5,395	156.0	7.5.	1 - 6 - 77	1.4
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- 18.5	Ч		<u></u>		L		ــــــــــــــــــــــــــــــــــــــ	ــــــــــــــــــــــــــــــــــــــ			 -

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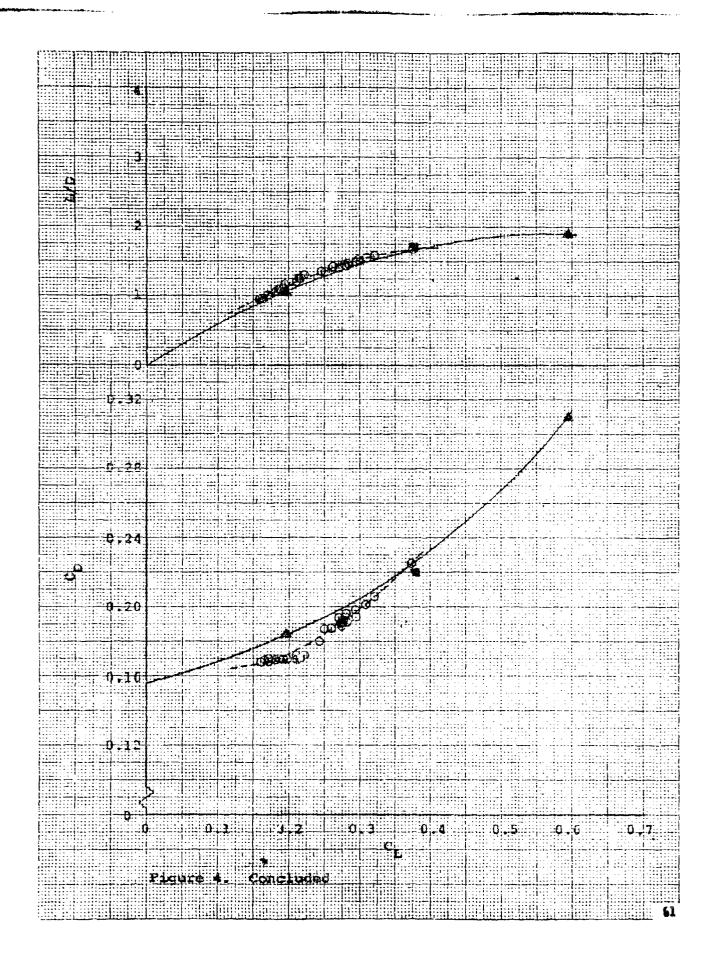
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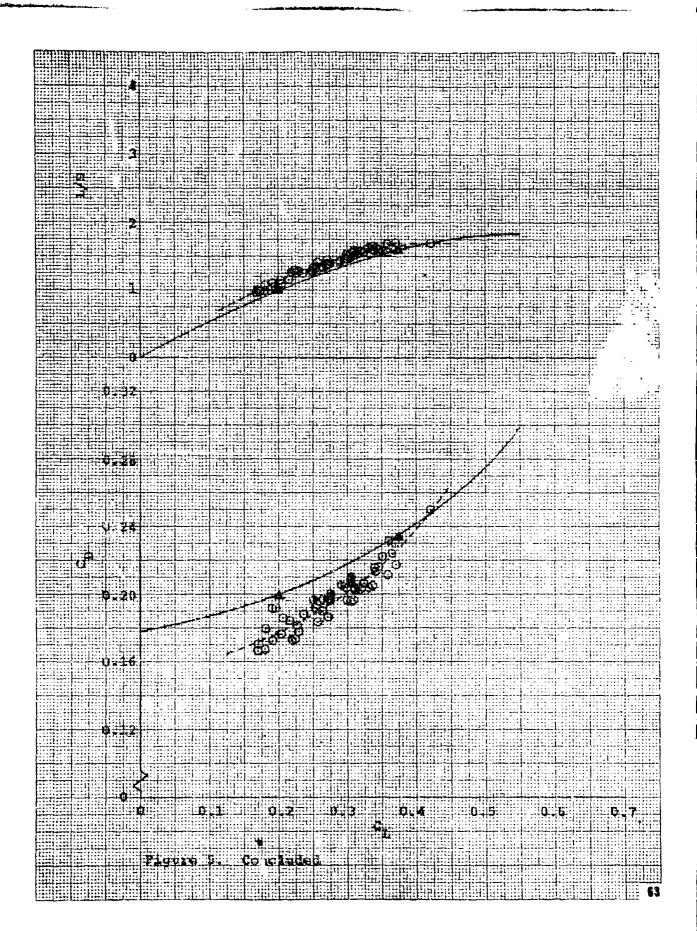
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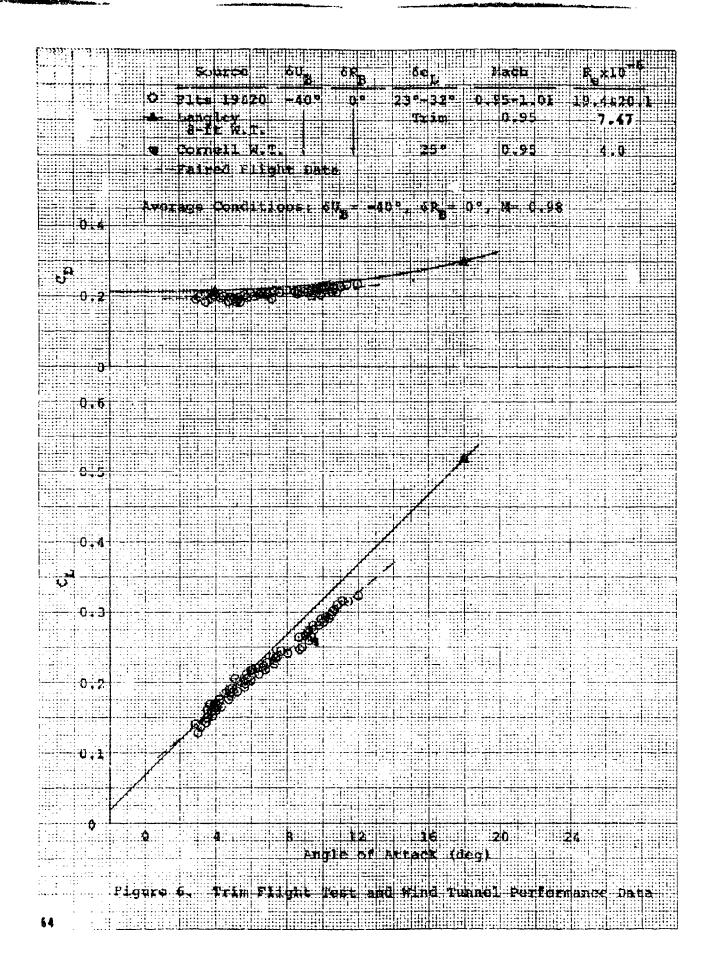
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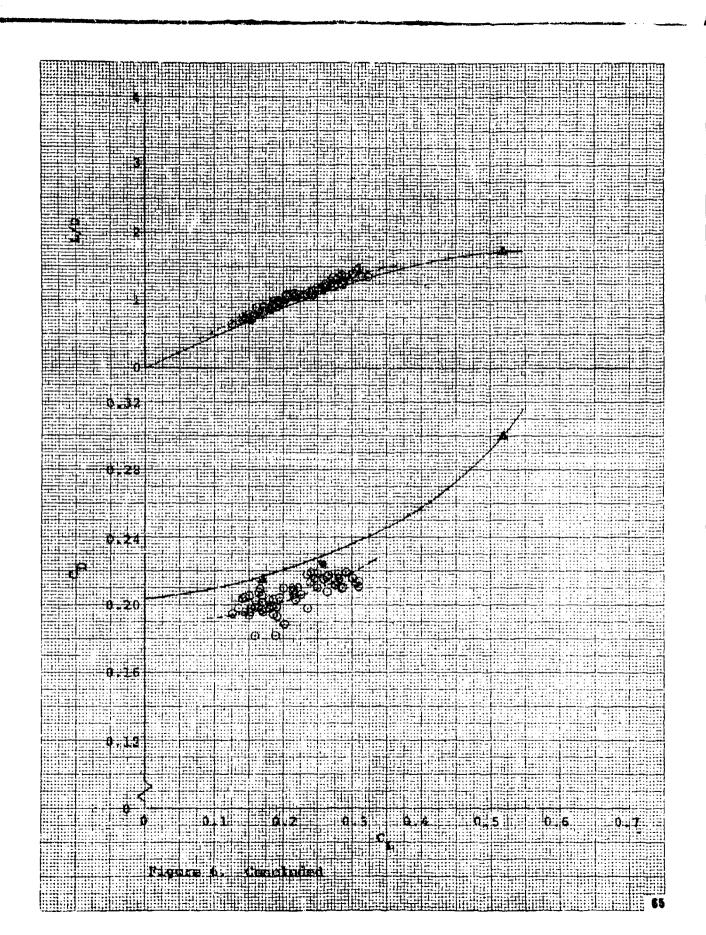
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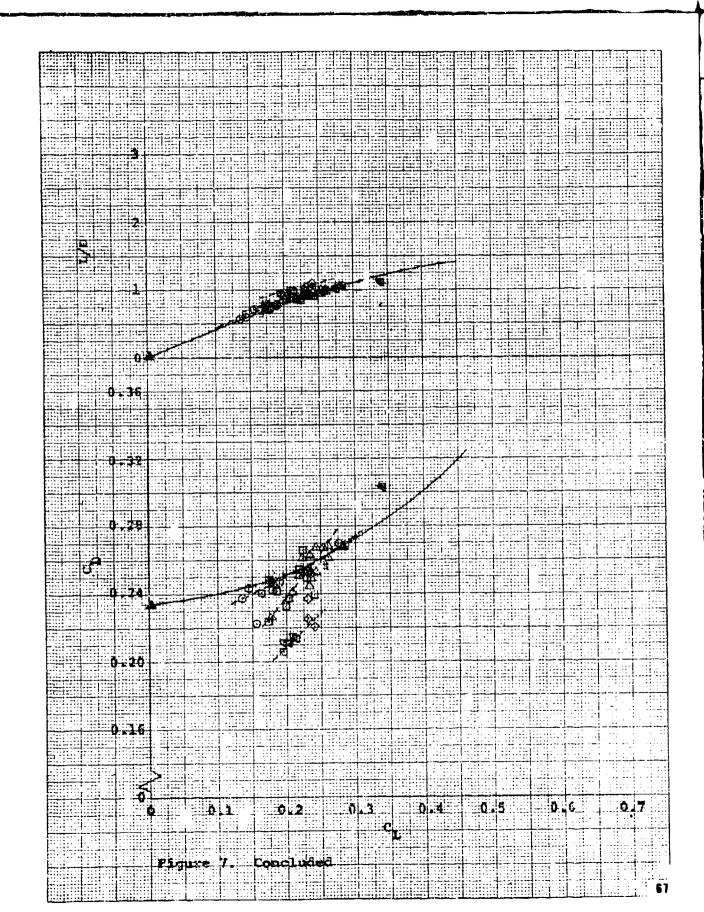
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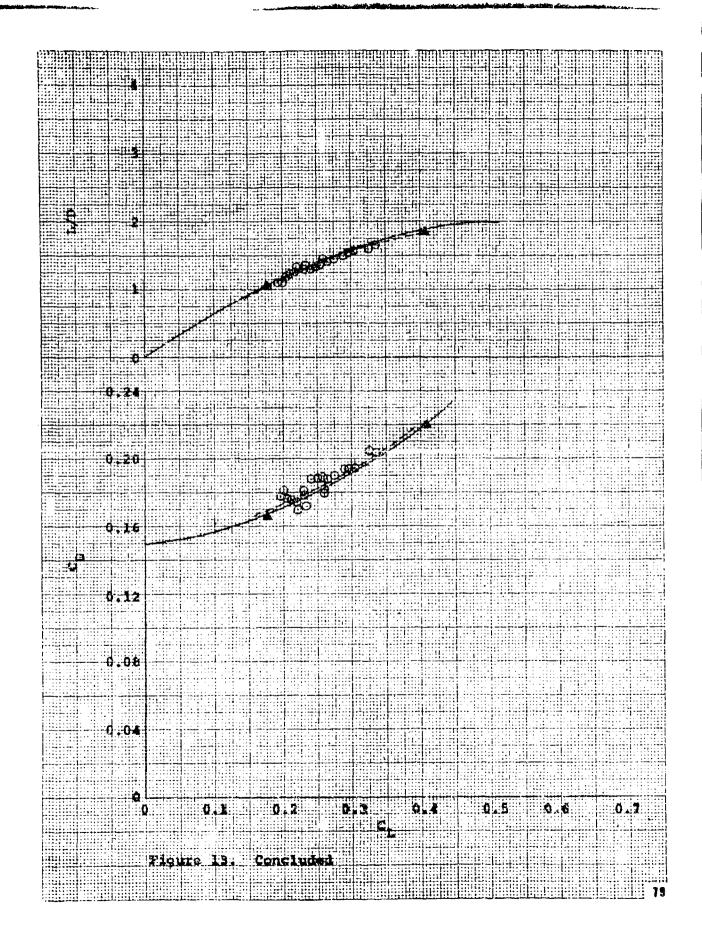
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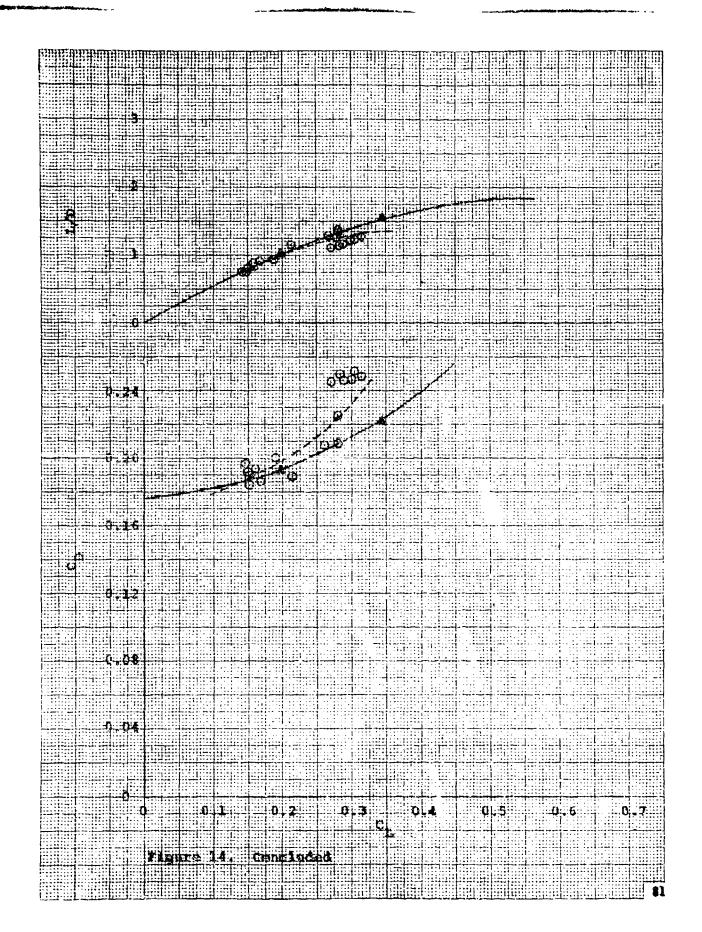
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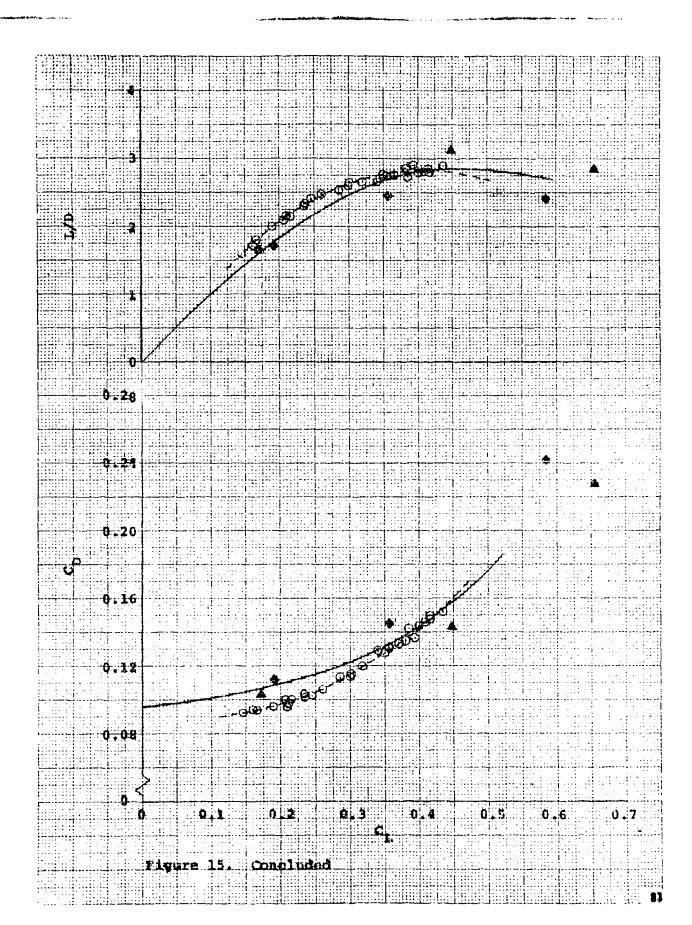
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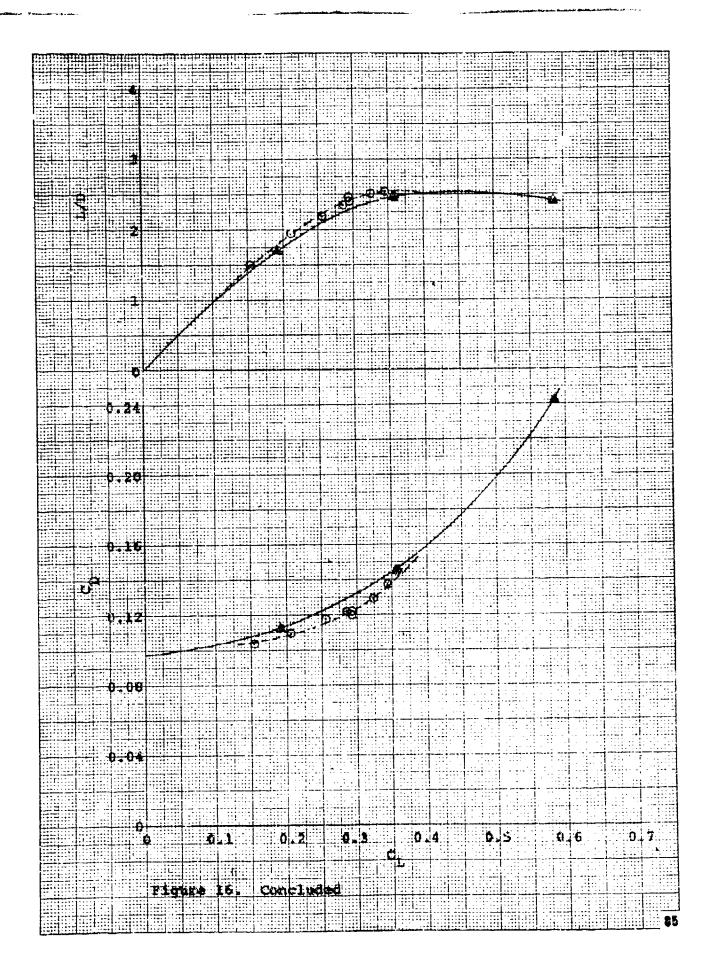
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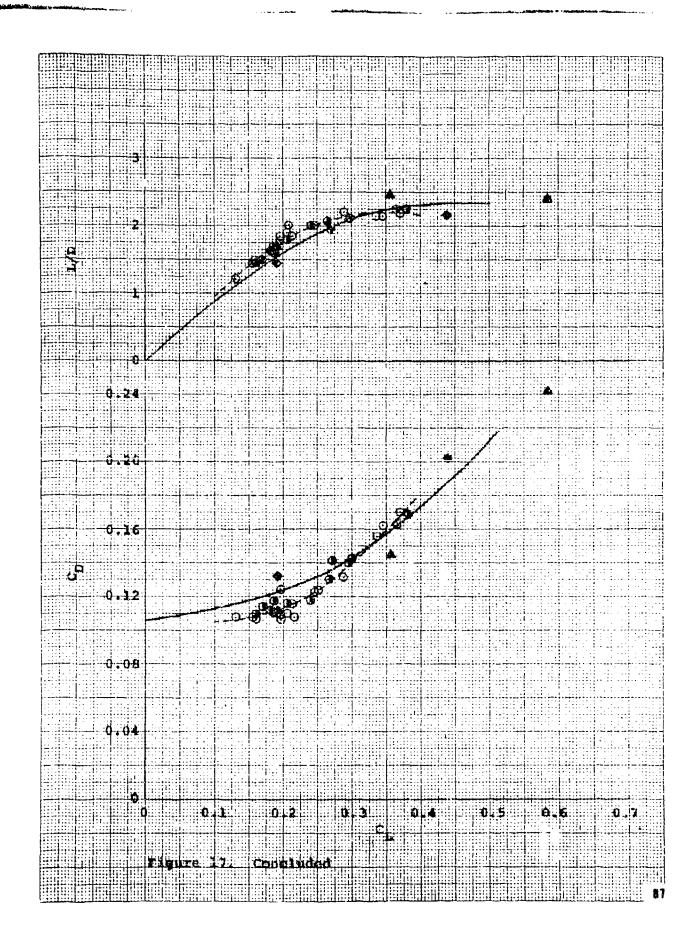
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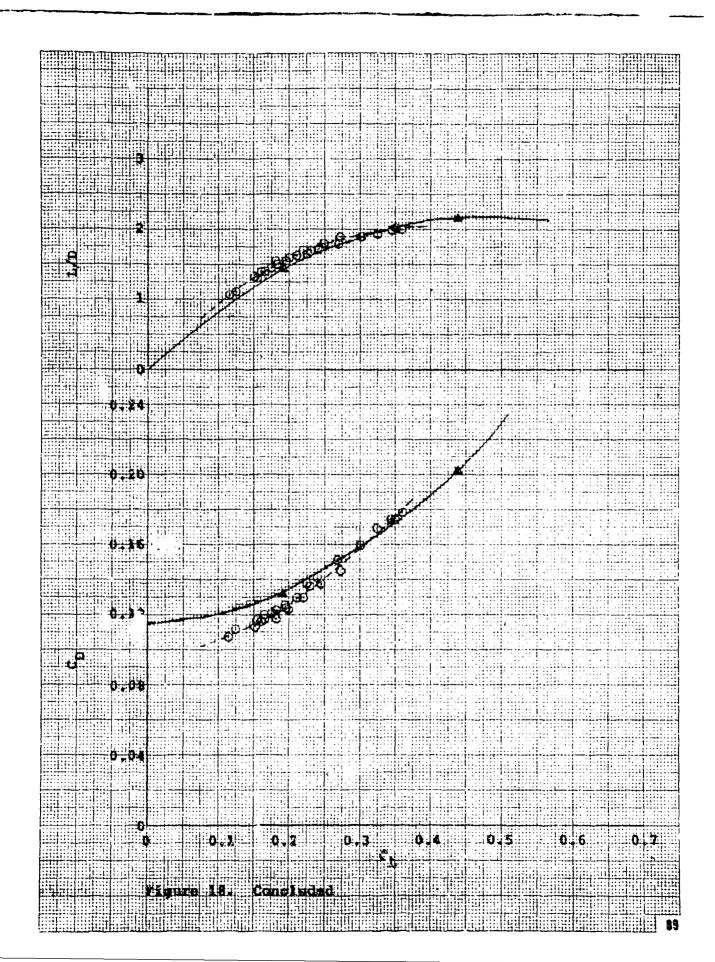
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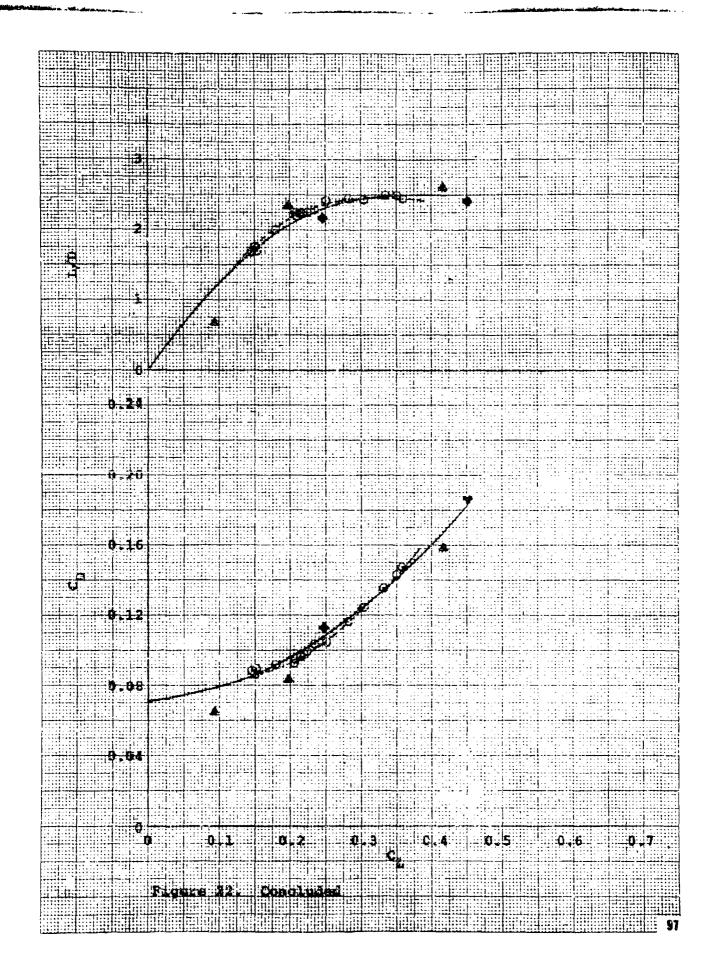
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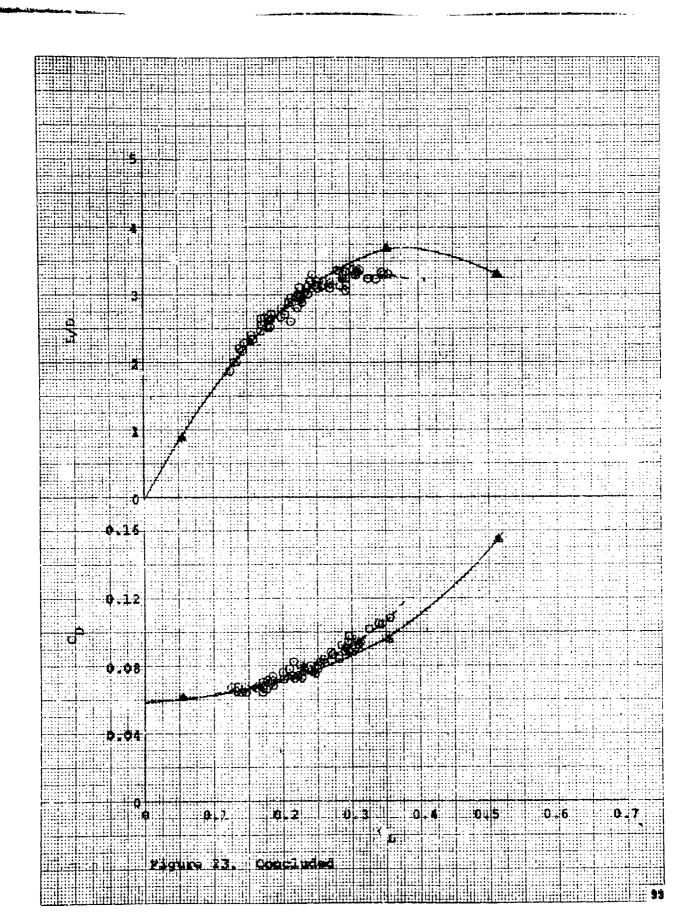
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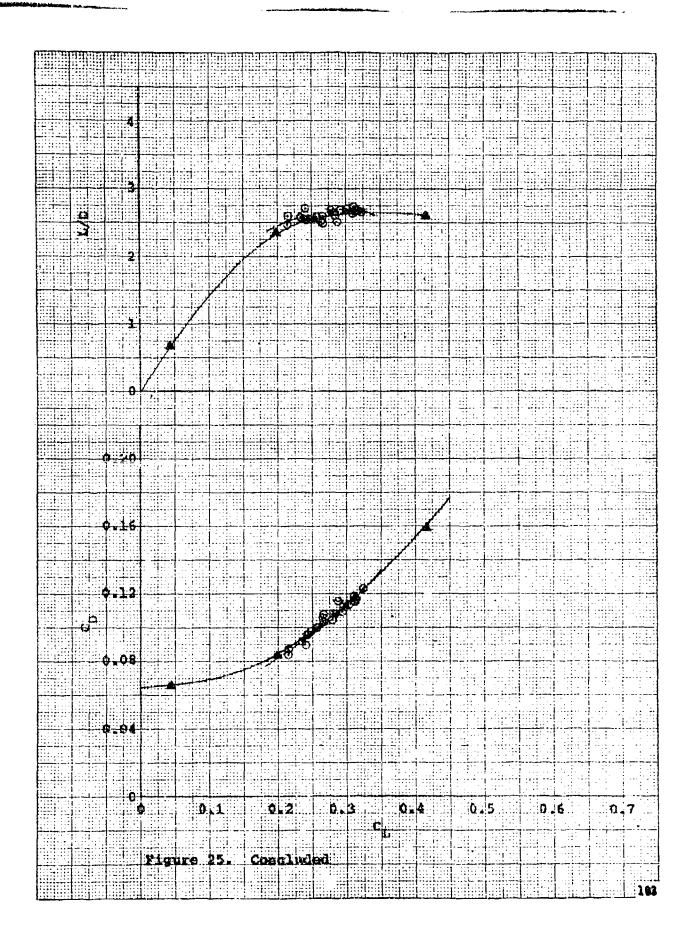


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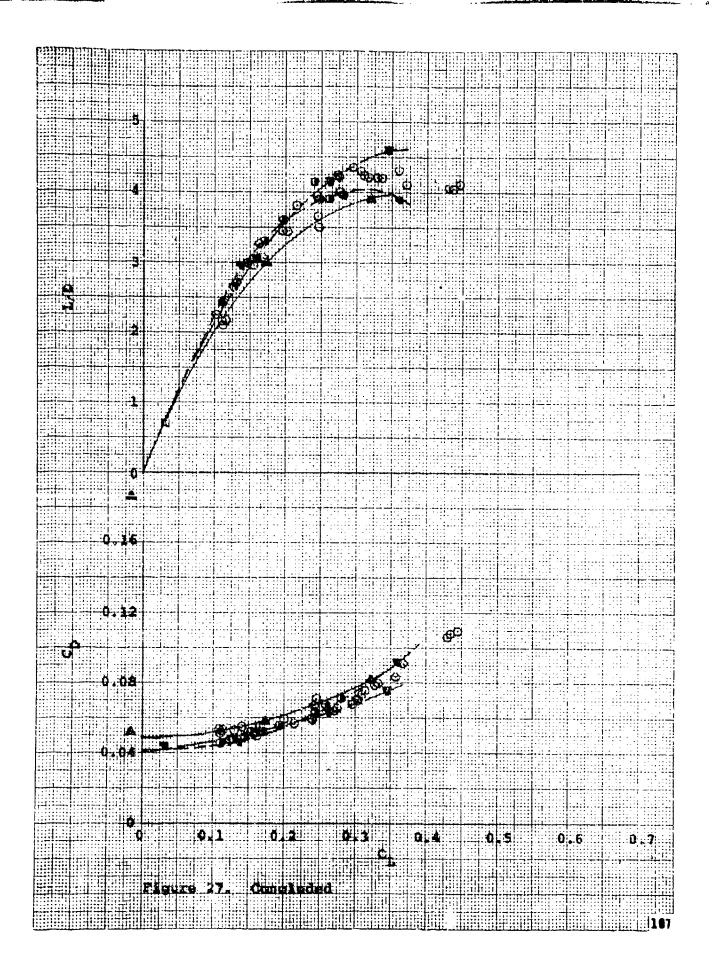
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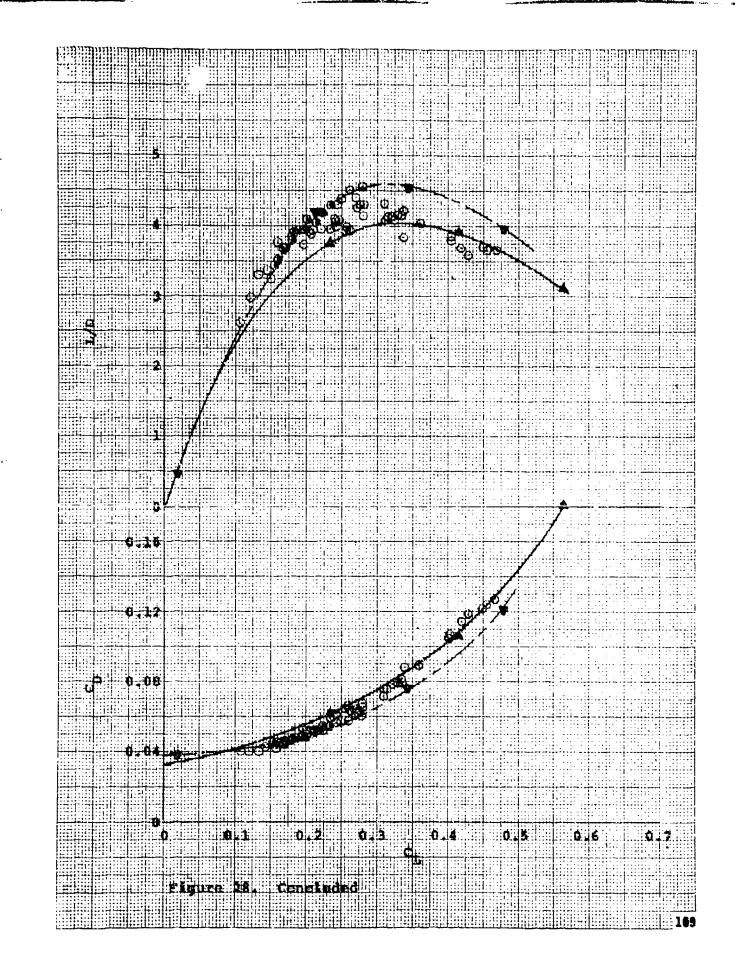
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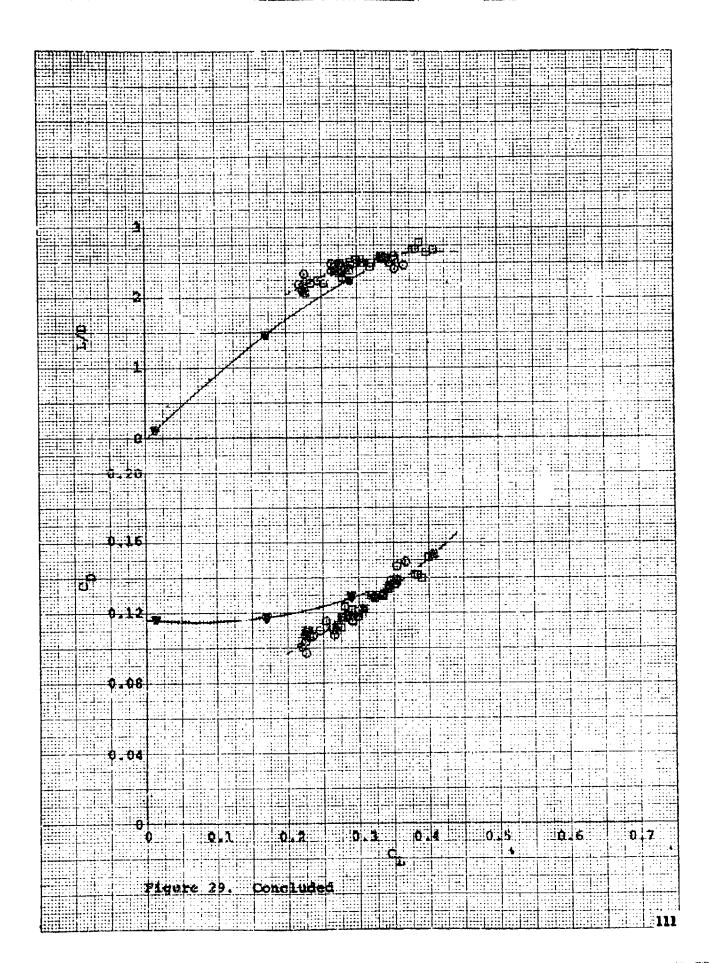
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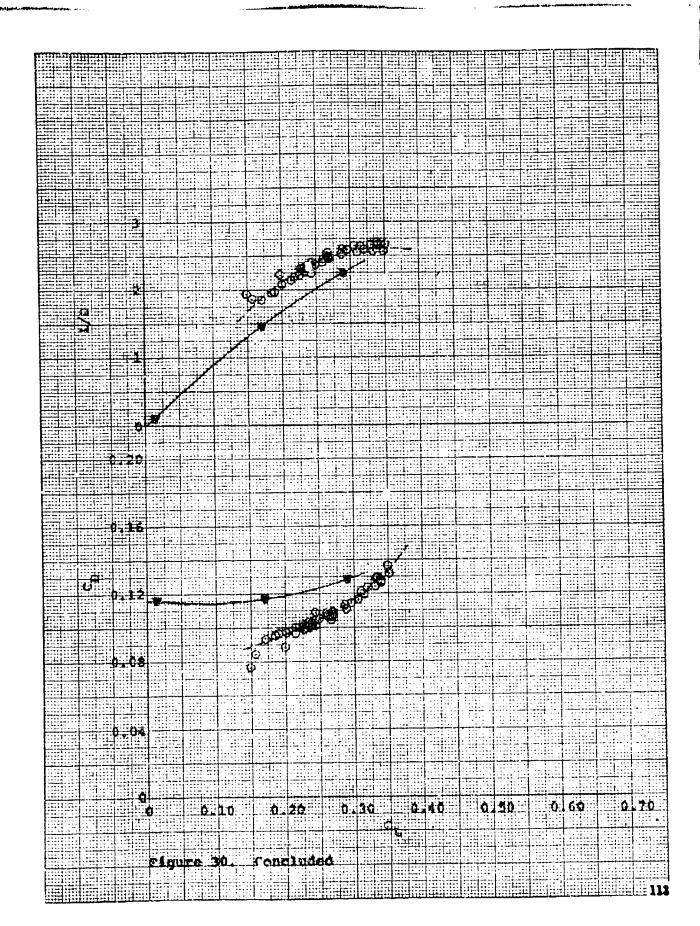
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APPENDIX II ANGLE OF ATTACK CORRECTIONS

Figure 1 shows the results of a calibration performed on the angle of attack vane attached to the nose boom in the full scale wind tunnel at a Mach number of 0.2. In an attempt to verify the wind tunnel results, data from the second flight are also presented. Flight data were obtained by computing flight path angle from radar data and subtracting it from the measured pitch angle to obtain a true angle of attack while the vehicle was at zero bank angle. This is not a very accurate technique due to wind effects and inherent inaccuracies in the radar data. The data, though scattered, substantiated the wind tunnel calibration of the X-24A angle of attack vane and established higher confidence in the wind tunnel results.

The nose boom was mounted at a 6-degree nosedown angle with respect to the longitudinal axis of the vehicle to minimize errors in the airspeed system pressure measurements. The true angle of attack as established by the wind tunnel (α_T) was defined by the relationship

$$a_{\rm T} = 0.87 \ (a_{\rm i}) = \frac{a_{\rm i}}{1.15} = \frac{a_{\rm boom} + 6 \ {\rm deg}}{1.15}$$

where $a_i = a_{boom} + 6 \text{ deg}$

From this equation

$$\Delta \omega_{\varepsilon} = \alpha_{1} - \alpha_{T} = \alpha_{1} - (0.87 \ \alpha_{1}) = 0.13 \ \alpha_{1}$$

at 0.2 Mach number.

To obtain the total upwash correction throughout the subsonic Mach number region, an attempt was made to compute the body and noseboom upwash effects using the methods of reference 10. The asymmetric shape of the X-24A made it very difficult to calculate the body upwash effects. Calculations were made using equivalent bodies of revolution generated individually by the top, bottom and side contours. Another calculation used the cross sectional area distribution of the X-24A to compute the radius of an equivalent body of revolution at a series of stations along the body. Both these methods yielded values for $\Delta\alpha_{\epsilon}/\alpha_{i}$ at 0.2

Mach number that were less than half the wind tunnel value, and so were discarded.

Finally the equation from the original wind tunnel curve,

$$\alpha_{\rm T} = \frac{(\alpha_{\rm boom} + 6 \text{ deg})}{1.15}$$

was modified to incorporate compressibility effects by applying a Mach function to the correction factor as follows:

$$\alpha_{\rm T} = \frac{(\alpha_{\rm bcom} + 6 \text{ deg})}{1 + .15 \sqrt{1 - M^2}}$$

This expression is plotted in terms of $\Delta\alpha_\epsilon/\alpha_i$ versus M in figure 2 along with the full scale wind tunnel point.

A correction was also made to angle of attack for the effects of boom bending due to normal accelerations ($\Delta\alpha_{bb}$). A final correction was made to α for pitch rate on the α vane which was 17.1 feet forward of the vehicle's center of gravity ($\Delta\alpha_{G}$).

Therefore, true angle of attack was obtained from the following summation:

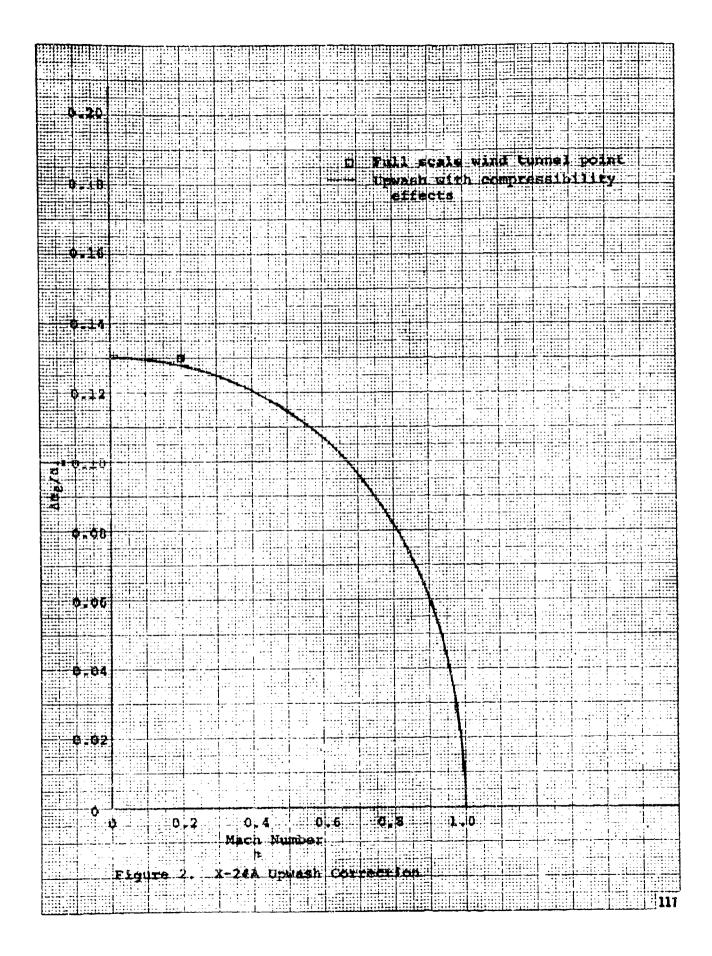
$$\alpha = \frac{\alpha_{boom} + 6 \text{ deg}}{1 + .15 \sqrt{1 - M^2}} + \Delta \alpha_{bb} + \Delta \alpha_{q}$$

where

$$\Delta a_{bb} = (.0666) (a_{z_b} - 1)$$

$$\Delta a_{q} = (17.1)(q/V_{t})$$

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APPENDIX III X-24A POSITION ERROR DETERMINATION

This appendix was adapted from an AFFTC Flight Test Technology Branch memo written by Christopher J. Nagy in January 1971.

INTRODUCTION

The position error measurement on the X-24A lifting body presented problems that are not encountered on most other types of aircraft. The common methods of position error measurements such as the tower fly-by, the ground speed course and the pacer would not work with the X-24A because there was little time spent in stabilized flight. The X-24's only power source was a rocket engine, which made it a boost-glide type vehicle. Thus, the aircraft was either gaining or losing energy very fast. This means that the altitude and/or airspeed was increasing or decreasing rapidly. These changes in altitude and airspeed introduced problems of lag in the static and dynamic pressure and transients in the position error. In addition, problems in the instrumentation were encountered. These problems and their solutions will be discussed in this appendix.

TECHNIQUE

The method of obtaining the position error is basically a comparison between the aircraft-measured static pressure and the actual static pres-

sure at a given altitude. The data was prepared in a $\frac{\mathrm{TP}}{q}_{c_{1,2}}$ form and a $\mathrm{TP}_{c_{1,2}}$

$$P_p = P_s - P_a$$

$$M_{pc} = M - M_{ic} = f \left(M_{ic}, \frac{T_{p}}{q_{c_{ic}}}\right)$$

from equation 5.65, page 74 of reference 6.

From equation 5.5, page 53, reference 6,

$$\frac{\mathcal{A}_{\mathcal{D}}}{q_{\mathcal{C}_{ic}}} = f \left(\mathcal{M}_{ic}, \mathcal{C}_{\mathcal{L}_{ic}} \right)$$

 c_{lic} effects were neglected so that $\frac{\mathrm{d} p}{q_{c_{\mathrm{lic}}}}$ is a function of instrument

corrected Mach number only. The aircraft measured a static pressure and a dynamic pressure using a standard NASA nose boom. The dimensions of this boom are given in figure 6. The actual static pressure was calculated from a Rawinsende barloon temperature versus altitude curve using the hydrostatic equation (equation 1.17, page 87).

Once the necessary parameters were obtained, the first step was to limit the usable portions of a flight to those where the climb or sink rate was less than 150 feet per second. This helped to eliminate the lag problem. The second step was to select times that gave $\rm M_{IC}$ values at approximately 0.005 increments between the minimum and the maximum $\rm M_{IC}$ values available within the usable portions of the flight. More spacing than 0.005 gave uncertain data; less than this gave unnecessary scatter. Once the times and $\rm M_{IC}$ values were obtained, the $\rm P_{A}$ values

were derived. First the altitudes at the specific times were recorded from an altitude versus time radar track. Then Pa values were obtained for each altitude from weather balloon data. The Pic counts from the vehicle were then recorded at the same times and two slightly different calibrations were used depending on whether the pressure was increasing or decreasing. (The pressure transducer transmitted digital counts to the ground and calibrations were used to obtain pressure.) This was

the ground and calibrations were used to obtain product done to account for hysteresis. Next, $\frac{\partial P}{\partial c}$ was calculated for each Mic ic

value and plotted versus M_{ic} . Once the $\frac{\Delta P_p}{q_{c_{ic}}}$ values were known, ΔM_{pc} was

found from the following equations (equations 5.63 and 5.65, page 67, reference 6) for ${\rm MM_{pc}}$ < .04:

$$M_{pc} = M - M_{ic} = \frac{(1.0 + 0.2 M_{ic}^2)}{1.4 M_{ic}} [(1.0 + 0.2 M_{ic}^2)^{3.5} - 1.0] \frac{M_{pc}^2}{q_{c_{ic}}}$$

for $M_{DC} > .04$:

$$\frac{\text{dP}_{p}}{\text{q}_{c_{ic}}} = \frac{\frac{1.4 \text{ M}_{ic} \text{ dM}_{pc}}{(1.0 + 0.2 \text{M}_{ic}^{2})} + \frac{0.7 (1.0 - 1.6 \text{ M}_{ic}^{2}) \text{ dM}_{pc}^{2}}{(1.0 + 0.2 \text{ M}_{ic}^{2})^{2}}}{(1.0 + 0.2 \text{ M}_{ic}^{2})^{3.5}} = \frac{\frac{1.4 \text{ M}_{ic} \text{ dM}_{pc}}{(1.0 + 0.2 \text{ M}_{ic}^{2})^{3.5}} + \frac{(1.0 + 0.2 \text{ M}_{ic}^{2})^{3.5}}{(1.0 + 0.2 \text{ M}_{ic}^{2})^{3.5}}}$$

This equation was solved to give a ${\rm AM_{pc}}$ versus M, curve.

RESULTS

The final data points of flights 10 to 13, 15 to 18, and 20, and the faired lines are shown in figure 1. Similar curves are given for comparison in figure 2 for the HL-10, another lifting body. Flights 1 through 9, the glide flights, were deleted for several reasons. First, they could not provide information on the higher portions of the Mach range, since, without power, the highest Mach number reached was about 0.7. Second, because the powered flights after burnout covered the same regime as the glide flights, the same data could be obtained from the powered flights. Third, the Rawinsonde balloon release time for flights 1 through 9 was not close enough to the launch time for the Pa values to be accurate. A late balloon release caused flight 14 to be deleted for the same reason. Flight 19 data was consistently high, and although a detailed study as to why was not made, a bad radar track was suspected.

Flight 21 data are not shown because it was consistently low. A closer look at the radar altitudes indicated that the radar may have lost track and may have been "hunting" during this time period. Flights 22 and 23, although not graphed, gave consistent data on or near the pre-

dicted curves. By flight 23, a fairly high level of confidence had been developed in the predicted curves. Succeeding flight data were not worked up since the Mach range was not extended beyond 1.35. A pressure system failure precluded use of data from flight 25 on which a maximum Mach of 1.6 was reached.

DISCUSSION OF RESULTS

The data indicate a leveling off of the $\Delta M_{\rm PC}$ curve below 0.55 $\rm M_{\rm iC}.$ It was decided to retain a straight line between 0.55 and 0.35 to accommodate the data between 0.40 and 0.55. Below 0.35, the position error was unimportant because the aircraft never flew below 0.35 Mach in the gear up configuration.

Transonic

The data indicated that the position error peaked around $\rm M_{i,C}=0.920$. A detailed look at airspeed and altitude traces showed that the dynamic pressure peaked at $\rm M_{i,C}=0.920$ and the static pressure at $\rm M_{i,C}=0.940$. From this information one might expect a curve such as the solid line shown in figure 3. On flight 18 almost no second break was visible, the static pressure jump being masked by corresponding changes in dynamic pressure. Flight 20 showed a more pronounced break at $\rm M_{i,C}=0.945$. To accommodate the flight data during the Mach jump, the $\rm \Delta M_{DC}$ curve was drawn bent slightly outward as shown by the dashed line (figure 3). This seemed to fit all data points mest accurately.

The data intercepted the $\Delta M_{\rm DC}$ zero line at about $M_{\rm IC}$ = 1.075. A position error of zero was used above Mach 1.075. Although plotted data points are slightly negative supersonically the values are felt to be well within the measuring accuracy of the system.

Correlation with Predicted Data

In April 1966, a study was made by the Martin Company in an effort to predict the position error curve for the X-24A. Dimensions from the existing nose boom were used in theoretical calculations to obtain ${\rm AP}_{\rm p}/{\rm q}_{\rm c_{ic}}$ as a function of Mach. This curve is shown in figure 4. Agree-

ment between predicted and flight measured values was generally good, and the agreement between $\rm M_{i\,C}=0.850$ and 0.750 was exceptional. The Mach number at which the break point occurs (where the bow wave passes over the pressure ports) was the same in each case.

Particular Problems Studied

Lag

Lag was studied for both the static and dynamic pressure systems and estimates were made for each. The static pressure lag can be calculated from the equation (reference 6, equation 4.2, 4.9, page 29, 30).

$$P_{lag} = \lambda \frac{dp}{dt}$$

Where
$$\lambda = \frac{32 \text{ mL}^2}{D^2 \text{ yP}} (1 + \frac{Q}{LA}) = \text{lag constant}$$

and m = coefficient of viscosity $\approx 3.0 \times 10^{-7}$ for X-24 temp range.

L = length of tubing, ft ≈ 8.93

D = diameter of tubing, ft $\approx 1.43 \times 10^{-2}$

 $_{\gamma}$ = ratio of specific heats ≈ 1.4

P = applied pressure, psf

Q = instrument volume, ft³, $\approx 1.447 \times 10^{-2}$

A = cross sectional area of tubing, ft² \approx 1.605 x 10⁻⁴

This leads to

$$\Delta P_{1ag} = \frac{29.7}{p} \frac{dp}{dt}$$

For a sink rate of 150 ft/sec at 40,000 ft

$$AP_{\text{lag}} = (29.7 \, \frac{\text{lb-sec}}{\text{ft}^2}) \, (3.9 \, \frac{\text{psf}}{\text{sec}}) \, (\frac{1}{391.6 \, \text{psf}}) = .3 \, \text{psf}$$

This was not enough to affect the validity of the results.

A similar study was made for dynamic pressure lag. Taking landing as the most extreme case since the rate of change in dynamic pressure is highest here.

$$q_c \approx 300 \text{ psf}$$

$$\frac{dq_c}{dt} \approx 15 \frac{psf}{sec}$$

 $\boldsymbol{\lambda}$ can be calculated similarly for the dynamic pressure system giving

$$aq_{c_{lag}} = (1.03 \times 10^2 \text{ sec})(15 \frac{psf}{sec}) = .155 \text{ psf}$$

Again this was not nearly enough to affect the results.

Zero Shifts in the Aircraft Pressure System

There was some concern that pressures taken at the balloon release time (within 30 minutes after landing) from both the balloon and the aircraft were different by as much as four psf after being corrected for altitude differences between the lakebed and the Rawinsonde station. The question was raised whether a zero shift correction should be applied to the X-24 pressure to make it conform with the ground pressure of the balloon. Starting with flight 14, checks were made on the two pressures and the results are shown in the following table:

Table I

DIFFERENCES IN X-24A AND WINTHER BALLOON STATIC PRESSURES

	Ground Pressu Ground Press	nsonde ure Minus X-24A sure Corrected ed Altitude	Total Change
Flight No.	Pre-flight Correction	Post-flight Correction	Between Pre- and Post-flight Checks
14	0.0	-2.9	2.9
15	+0.4	-2.0	2.4
16	+1.1	-1.6	2.7
17	+1.0	-1.5	2,5

It can be seen that the pre- and post-flight corrections were of opposite signs. The discrepancy was attributed to hysteresis in the X-24A static pressure transducer. Hysteresis, according to the manufacturer of the instrument is 7 psf. In actual calibration with pressure increasing and decreasing, the apparent hysteresis was more like 3 psf, which is in accordance with the change in pre- and post-flight checks. It was therefore concluded that the data should not be corrected for zero shifts but that hysteresis effects were of greater importance and should be accounted for when applying the calibrations.

Time Shifts

The most important factor in reducing the data was to insure that the balloon passed through the flight altitude at approximately the same time of day that the flight occurred. Best results were obtained when the balloon reached flight altitude within 30 minutes of flight time. Fl. Jht 14 is a good example of erroneous data because this rule was not observed. Due to transmitting difficulties, the balloon was not at altitude until four hours after the flight. Figure 5 shows the data according to the late balloon; in addition, data from a balloon seven hours prior to the flight and the expected results are shown. Data from late balloons may give good results depending on the stability of the atmosphere during that time period. However, any attempt at correcting for the time shift based on ground pressures is useless, since, while the pressure at ground level may have changed 1 psf, the pressure at 40,000 feet may have changed 3 or 4 psf.

Pacer

The first flight with the rockets called for a powered profile which would keep the X-24A at a somewhat constant altitude and speed for a reasonable amount of time. Therefore, this opportunity was taken to use a well calibrated T-38 pacer to establish some good points on the position error curve.

The pacer pilot turned on the event light three times during the flight indicating that he felt he was stabilized alongside the X-24A. One point was taken during the powered portion at 35,000 feet and at an X-24A $\rm M_{IC}$ of 0.660. The other two points were taken in the pattern at test indicated Mach numbers of 0.426 and 0.419.

Indicated altitude, velocity, and temperature data were taken from the pacer photopanel and instrument corrected. This data was put into a computer program along with corresponding X-24A data and the pacer position error curve. The program computed position errors using a $\rm \Delta H_{PC}$ method and a $\rm \Delta V_{PC}$ method.

The results show good agreement between the two methods for the point at 0.660 Mach number. Values for $^{\Delta P}p/q_{_{C_{i,C}}}$ were 0.058 and 0.062

which correspond to AM's of 0.023 and 0.025. These values were close to the other data which gave a AM $_{\rm DC}$ of 0.029 at 0.660 M $_{\rm iC}$.

The two methods did not show good agreement for the two points in the pattern. Values for $\Delta P_p/q_{c_{i,c}}$ ranged from 0.115 to 0.067 correspond-

ing to $\Delta M_{\text{DC'S}}$ of 0.027 to 0.015. However, all points fall within the scatter of the other data.

The pacer points give good support to the data obtained through the other method explained in the report.



Figures 1 Through 6

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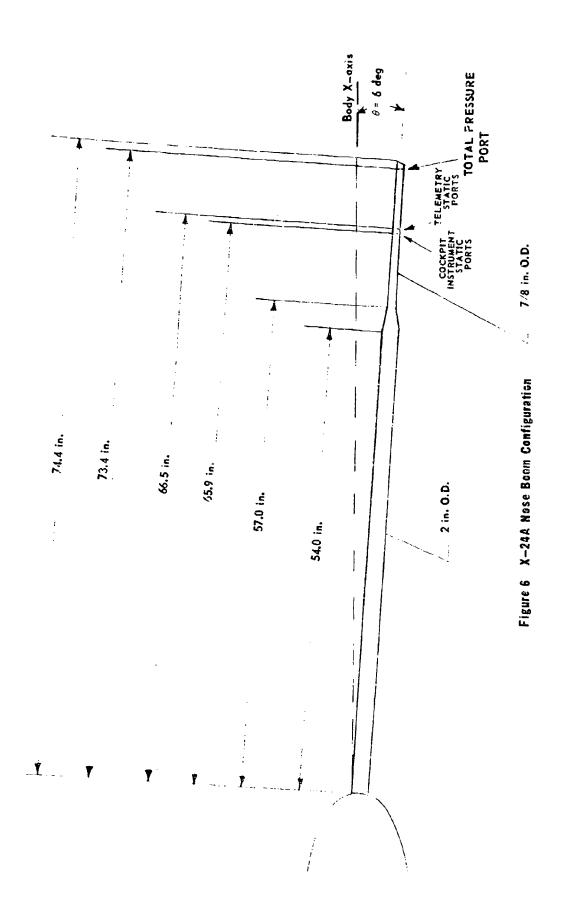
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APPENDIX IV DIGITAL COMPUTER PROGRAM FOR DETERMINATION OF PERFORMANCE CHARACTERISTICS

This computer program was written by David F. Richardson of the Flight Test Technology Branch.

An IBM 1620 digital computer program was used to correct flight data and compute performance characteristics of the X-24A. A computer listout of the program is shown in figure 1. A list of equations used in the program is also presented. The following corrections were made to the flight data by the program:

- 1. A correction to the accelerometers for a pitch misalignment angle ϵ with the X- and 2-axes.
- A correction for displacement of the accelerometers from the test center of gravity.
- 3. A correction to trim elevator position for pitch rate and for cg variation from a wind tunnel reference of 57 percent. This reference cg was used as the standard cg in the computer program. During the pushover/pullup the pitch rate was sustained by a δe input which contributed to lift and drag. Therefore, it was necessary to know the derivatives C_{m_q} , $C_{m_{\delta e}}$, $C_{c_{\delta e}}$, and $C_{N_{\delta e}}$, in order to take out the normal and chord force increments due to the δe increment required to maintain the pitch rate. From pitch rate (q), C_{m_q} , and $C_{m_{\delta e}}$, it was possible to determine the δe increment required to maintain the test pitch rate. The normal and chord force increments associated with this δe were subtracted from the test values of C_N and C_C to obtain the total test coefficients. The test values were then added to ΔC_N and ΔC_C due to variation from the standard cg to obtain the total standard coefficients for zero pitch rate at 57 percent cg.

Values of $C_{\rm L}$ and $C_{\rm D}$ at 57 percent standard cg were computed along with standard L/D by rotating the body axis coefficients using true angle of attack. The program printed out corrected trim elevator position, $C_{\rm N}$ and $C_{\rm C}$ test, and standard values for $C_{\rm N}$, $C_{\rm C}$, $C_{\rm L}$, $C_{\rm D}$, and L/D. The longitudinal stability and control derivatives, Mach number, and angle of attack, which were input data were also printed out for each data point. A sample printout is shown in figure 2. Table I gives a list of all symbols used in the program.

```
L/D DETERMINATION
C
      TYPE 1
                (//25X,18H L/D DETERMINATION/)
    1 FORMAT
      READ 3
    3 FORMAT (20% 27H A/C TYPE FLIGHT NO. DATE, 20%)
      TYPE 3
      TYPE 25
                (//48H TEST CN AND CA CORRECTED TO STANDARD CG AND (1=Ø)
   25 FORMAT
      TYPE 26
   26 FORMAT(//2X7H MN/CNT,4X1ØH ALPHA/CAT,3X9H CNDE/CNS,3X9H CADE/CAS)
      TYPE 27
   27 FORMAT (40x, 10x 9H CMDE/CLS, 3X 8H CMQ/CDS, 2X 11H DEC/(L/D)S//)
      READ 2, T, WT, TMAC, SAREA, E, X, Z
      READ 2, DXCG, DZCG, DXAX, DYAX, DZAX, DXAZ, DYAZ, DZAZ
      CK=1.7453292E-02
      E=E*CK
   10 READ 2,HT,P,R,DP,DR
READ 2,CNDE,CADE,CMDE,CMG,DE
      READ 2,MN,OB, VT, ALPHA, AX, AZ, O, DO
    2 FORMAT (8F10.0)
       VT=VT*1.688
      AXC=AX*COS(E)-A7.*SIN(E)
      AZC=AZ*COS(E)+AX*SIN(E)
      AG=32.175-(3.086E-06)*HT
      AXCG=AXC+(CK*(DYAX*DR-DZAX*DQ)+CK**2*DXAX*(Q**2+R**2))/AG
      AZCG=AZC-(CK*(DXAZ*DQ-DYAZ*DP)+CK**2*DZAZ*(Q**2+P**2))/AG
      CNT=AZCG*WT/(OB*SAREA)
      CAT=(T-WT*AXCG)/(QB*SAREA)
      DDET= CMO*O*TMAC*CK/(CMDE*2.Ø*VT)
      CNTC=CNT+DDET*CNDE
       CATC=CAT+DDET*CADE
      DCMCG=(DXCG*CNTC-DZCG*CATC)/TMAC
      DDES=-DCMCG/CMDE
       CNS=CNTC+DDES*CNDE
       CAS=CATC+DDES*CADE
       ALPH1=ALPHA*CK
       CLS=CNS*COS(ALPH1)-CAS*SIN(ALPH1)
       CDS=CAS*COS(ALPH1)+CNS*SIN(ALPH1)
       SLDR=CLS/CDS
       1F(Z) 21,21,20
   20 DDET=DDFT/CK
       DDES=DDES/CK
   21 DEC=DE+DDET+DDES
      TYPE 29,MN,ALPHA,CNDE,CADE,CMDE,CMO,DEC TYPE 29,CNT,CAT,CNS,CAS,CLS,CDS.SLDR
   29 FORMAT (F9.6, F13.6, 5F12.6)
       TYPE 30
   3Ø FORMAT (/)
       IF(X) 15,15,10
       END
          :OTE:
                  *Indicates multirlication.
```

Figure 1 Performance Digital Computer Program Listout

**Indicates an exponent.

Correction to Accelerometers for itch Misslignment &

$$a_{x_c} = a_{x} \cos \varepsilon - a_{z_b} \sin \varepsilon$$

$$a_z = cos \varepsilon + a_x sin \varepsilon$$

Altitude Correction to Acceleration due to Gravity

$$a_g = 32.175 - (3.086 \times 10^{-6}) \text{ ii}$$

Correction to Accelerometers to Test cg

$$a_{x_{cg}} = a_{x_{c}} + \frac{(\Delta y_{ax} \cdot \dot{r} - \Delta z_{ax} \cdot \dot{q}) + \Delta x_{ax} (\sigma^{2} + r^{2})}{a_{g}}$$

$$a_{z_{cg}} = a_{z_{c}} - \frac{(\Delta x_{az} \cdot q - \Delta y_{az} \cdot p) - \Delta z_{az} (\sigma^{2} + p^{2})}{a_{g}}$$

where

 $+\Delta x = distance$ of accelerometer forward of test cg in ft

+Ay = distance of accelerometer right of test og in ft

 $+\Delta z$ = distance of accelerometer down from test cg in ft

subscripts ax - longitudinal accelerometer

az - normal accelerometer

Calculation of $C_{\widetilde{I}\widetilde{I}}$ and $C_{\widetilde{C}}$ from Corrected Accelerations

$$c_{N_{T}} = (a_{z_{cg}} \cdot W)/(a_{c} \cdot S)$$

$$C_{c_{\underline{T}}} = (\underline{T} - \underline{W} \cdot \underline{a}_{\underline{X}_{\underline{C}\underline{S}}})/(\underline{q}_{\underline{c}} \cdot \underline{S})$$

Correction to Elevator for Zero Pitch Pate

$$\Delta \delta e_t = c_{m_q} \cdot q \cdot \overline{c} / c_{m_{\delta e}} \cdot 2V_t$$

Correction to C_{χ} and C_{ζ} for Elevator Trim Change

$$C_{N_{TC}} = C_{N_{T}} + \Delta \delta e_{t} \cdot C_{N_{\delta e}}$$

$$C_{c_{\underline{n}}c} = C_{c_{\underline{n}}} + \Lambda \delta e_{\underline{t}} \cdot C_{c_{\underline{\delta}e}}$$

Change in Pitching Moment Induced by Translating Test og to a 57

Percent Standard eg

$$\Lambda C_{m_{eg}} = (\Delta x_{eg} \cdot C_{m_{Te}} + \Delta z_{eg} \cdot C_{c_{Te}}) / \bar{c}$$

where

+Ax $_{\text{CF}}$ = distance of test og forward of 57% standard og in ft

 $+\Delta z_{cc}$ = distance of test cg down from 57% standard cg in ft

Correction to Elevator to Froduce Zero Pitch Nate at Standard og

$$\Delta \delta e_{s} = -\Delta C_{m_{og}}/C_{m_{\delta e}}$$

Correction to $C_{\frac{1}{2}}$ and $C_{\frac{1}{2}}$ for Elevator Trim Change to Standard og

$$C_{ij} = C_{ij} + \Delta \delta e_{s} \cdot C_{ij} \delta e$$

$$C_c = C_{c_{Tc}} + A\delta e_{s} \cdot C_{\delta e}$$

Calculation of C $_{\rm L^{\bullet}}$ C $_{\rm D}$ and L/D at 57 Percent Standard cg

$$C_L = C_{ii} \cos \alpha - C_{c} \sin \alpha$$

$$C_D = C_C \cos \alpha + C_H \sin \alpha$$

$$L/D = C_L/C_D$$

Calculation of Total Elevator Deflection Corrected to Zero Pitch Rate at 57 Percent Standard og

$$\hat{o}e_c = \delta e + \Delta \delta e_t + \Delta \delta e_s$$

L/D DETERMINATION

X-24A

X - 3 - 5

21 AUG 69

TEST ON AND CA CORRECTED TO STANDARD OF AND OF

MN/CNT	ALPHA/CAT	CNDE /CNS	CADE/CAS	CMDE/CLS	CMQ/CDS	DEC/(L/D)S
.487000	9.010000	.410000	. 074000	110000	358000	3.518786
.314289	.023167	.301042	. 020776	.294074	.067665	4.345976
.488000	9.300000	.415000	. 072 000	110000	359000	3.728989
.330789	.024566	.315933	. 021 989	.308227	.072756	4.236444
.499000	9.120000	.410000	.073000	110000	359000	3.265271
.324826	.024363	.310338	.021783	.302962	.070697	4.285311
.491000	9.150000	.410000	. 0731100	110000	360000	3.412839
.321896	.028y72	.307890	. 026478	.299762	.075102	3.991389
.491000	9.330000	.417000	.072690	110000	360000	2.204203
.319780	.025588	.305837	.023180	.298033	.072456	4.113252
.491000	10.070000	.425000	.071000	110000	- 360000	.981107
.350825	.019234	.337036	.016931	.328883	. 075601	4.350223
.492000	12.960000	.450000	.066000	109000	360000	.393309
.419277	.011253	.401553	.008653	.390882	.092361	4.232091
.492000	13.740000	.467000	. 062000	108000	360000	882719
.456329	.003263	.435197	. 000457	.422634	.103811	4.071185
.491000	14.110000	.470000	. Ø6ØØØØ	107000	360000	-1.973856
.472640	.001601	.448656	ØØ1459	.435476	.107959	4.033700

Figure 2 Sample Output from Performance Digital Computer Program

Table I
PERFORMANCE DIGITAL COMPUTER PROGRAM SYMBOLS

Quantity	Program Symbol	Units
îhrust	т	1b
Vehicle weight	WT	1b
Mean aerodynamic chord	TMAC	£t
Wing area	SAREA	ft ²
Dynamic pressure	QB	lb/ft ²
Mach number	MN	
Corrected altitude	HT	ft
True airspeed	VT	kt
True angle of attack	ALPHA	deg
Longitudinal acceleration	XA	g's
Normal acceleration	A2	g 's
X Distance to test cq	ĐXCG	ft
7 Distance to test og	D7.CG	ft
Distances of longitudinal	DXAX	ft
accelerometer from test og	DYAX	ft 1t
Distances of normal	DXAZ	ft.
accelerometer from	DYAZ	f t
test cg	SAZG	ft
Poll rate	P	deg/sec
Pitch rate	Ç	deg/sec
Yaw rate	Р	deg/sec
Roll acceleration	DP	deg/sec ²
Pitch accoloration	ÇQ	deg/sec ²
Yaw acceleration	DP	deg/sec ²
CN 5e	CNDE	deg ⁻¹ or rad
c _{Ase}	CADE	deg ⁻¹ or rad
С _{т.} е	CMDE	deg ⁻¹ or rad ⁻
C _{Fa}	CMO	rad-1
Accelerometer alignment angle	Ξ	deg
Conversion constant	Ck	rad/deg
Accelerations corrected for alignment	AZC AXC	g's g's
Accelerations corrected to zero rotation rates	AXCG AZCG	rad/sec ² rad/sec ²
Test normal force coefficient	CNT	
Test axial force coefficient	CAT	
Elevator correction for pitch rute	DDET	deg
Test normal and axial force coefficients corrected to zero pitch rate	CNTC CATC	
eg Correction to pitching moment	DOMOG	
gg Correction to elevator	DUES	deg
Standard normal and axial coefficients	CNS CAS	
Standard lift and drag coefficients	CLS CDS	
Standard lift to drag ratio	SLDR	
Elevator positi	DE	deg
Corrected elecator position	DEC	deg

APPENDIX V TIP FIN FLOW SEPARATION STUDY

This appendix was adapted from an AFFTC Flight Test Technology Branch memo written by Robert G. Hoey in January 1970 after flight eight of the test program.

INTRODUCTION

Flow separation over the inner tip fin and rudder surfaces (tip fin stall) has occurred on each of the eight X-24A glide flights. The existence of separated flow is apparent in the rudder hinge moment and accelerometer data. It has been verified by tuft photos taken from the center fin camera and from chase plane photos. This has been manifested to the pilot as a mild, high frequency, Mach-type buffet.

TEST DATA

Tuft photos of the right tip fin from the center fin camera for 3 flight conditions are shown in figure 1. Three typical rudder hinge moment time histories are shown in figure 2 as the flight conditions crossed the tip fin stall boundary. These hinge moment traces are for the right rudder surfaces. The left rudder traces are generally similar, but the transition between attached and separated flow appears to be more gradual on the left side. Correlation between tuft photos and hinge moment traces for the onset of separation is good.

DISCUSSION

Analysis of wind tunnel data showed a large variation in the derivatives $C_{\rm nz}$ and $C_{\rm lz}$ for the Mach/angle of attack region in question (figure 3). The point of nonlinearity was probably related to the stalling of the tip fin. Notice in figure 3 that increasing the upper flap setting tended to increase the level of static stability in the region of separated flow but did not alter the angle of attack at which the flow separation occurred. Wind tunnel data at other Mach numbers showed similar trends and the apparent stall angle of attack has been plotted versus Mach number in figure 4. The flight conditions for the tuft pictures are shown by the three numbered circles. The three hinge moment time histories are represented by the arrowed lines. Reasonable correlation can be seen between the wind tunnel stall boundary and the flight test data points.

A summary plot of all observed separation or reattachment occurrences is shown in figure 5. These points represent a variety of upper flap and rudder bias configurations ($\delta U_B = -21$ degrees, -23.5 degrees, -30 degrees; $\delta R_B = 0$ degrees, -5 degrees, -10 degrees). Although there is a considerable amount of scatter in the plotted data, efforts to separate out the possible effects of sudder bias, upper flap bias, side-slip angle, rudder trim, or right versus left tip fin stall have been unsuccessful. It does appear that the intensity of the hinge moment vibrations was less in the stall region at the higher upper flap settings, a trend similar to that observed in the wind tunnel variations of $C_{\rm hi}$ and $C_{\rm hi}$ (figure 3).

The tip fin stall probably occurred when the local flow near the fin leading edge reached sonic velocity. This is indicated by the strong dependence of the stall boundary on Mach number and angle of attack.

A two-dimensional flow analysis for the tip fins conducted by R. Banner, RASA Flight Research Center, using wind tunnel pressure data confirms the possibility of local Mach numbers approaching 1.0 at freestream Mach numbers as low as 0.6.

EFFECT ON FUTURE FLIGHTS

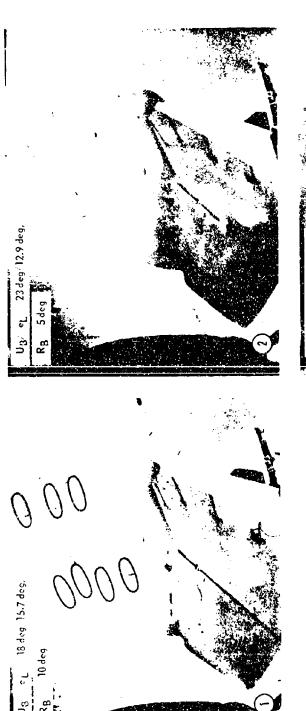
The tip fin stall observed during test flights to date has not produced any noticeable degradation in the inherent aerodynamic stability or controllability of the X-24A. The flow separation has, however, always occurred asymmetrically (or with asymmetric intensity) such that steady state pilot lateral control inputs of as much as 20 degrees have been required to maintain wings level flight. Since the lateral control power is quite low this could produce a dangerous situation if the asymmetry becomes worse at higher Mach numbers. Rudder trim has been used very effectively to compensate for a steady lateral mistrim; however, momentary angle of attack or configuration changes near the stall boundary produced transient lateral motions which were quite annoying to the pilot.

Analysis of the wind tunnel data before the first flight produced the boundary shown by the dotted lines in figure 5. This boundary was based upon analysis of the upper flap lateral control data which shows a sharp drop in effectiveness for flight conditions above this line. Note in tuft photo No. 2 that the flow over the upper flap is still attached even though the tip fins are stalled. This flight condition is above the predicted tip fin stall boundary, but below the predicted upper flap stall boundary. It is therefore anticipated that the predicted upper flap stall boundary is probably still valid with consequences which may be more severe than those associated with the tip fin stall. Since this boundary does appear to be a function of upper flap position larger upper flap settings should be used during exploratory flights to higher Mach numbers.

CONCLUSIONS

The tip fin stall observed on X-24A glide flights appears to correlate with nonlinearities in certain wind tunnel derivatives and is not in itself cause for concern. The flow separation boundary is most likely associated with the occurrence of sonic velocity near the tip fin leading edge and appears to be a primary function of only Mach number and angle of attack. Lateral-directional trim requirements associated with asymmetric tip fin stalling (or asymmetric intensity of the flow separation) was annoying to the pilot and could have become dangerous if the asymmetry became more severe at higher Mach numbers. Predicted boundaries for flow separation over the upper flaps are probably still valid and should be considered in the planning of high Mach number flights.

¹Flights subsequent to this memo did not invalidate these statements.



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TIP FIN FLOW

Attached Separated Attached

4.3 deg 12.4 deg 9.5 deg

.447 .577 .493

PHOTO

Figure 1 X-24A Center Fin Camera Tuft Photos

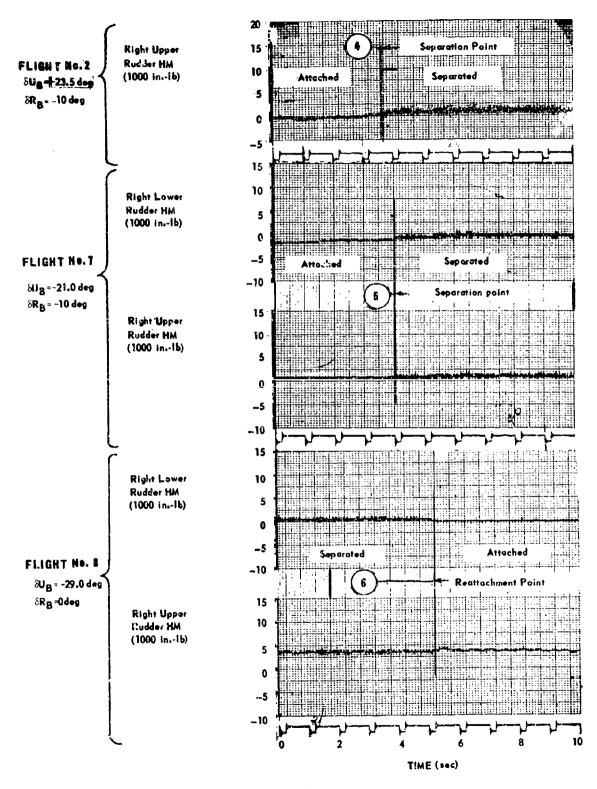


Figure 2 X-24A Ruddor Hinge Memonts

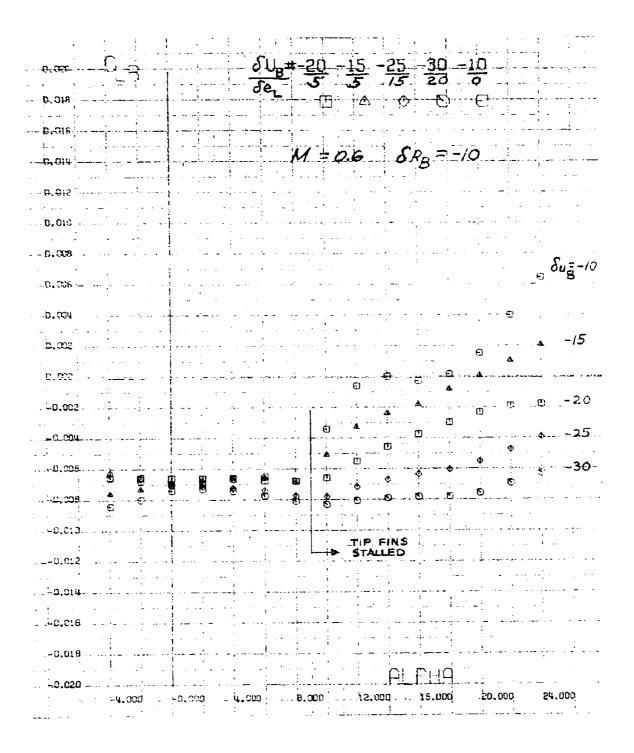
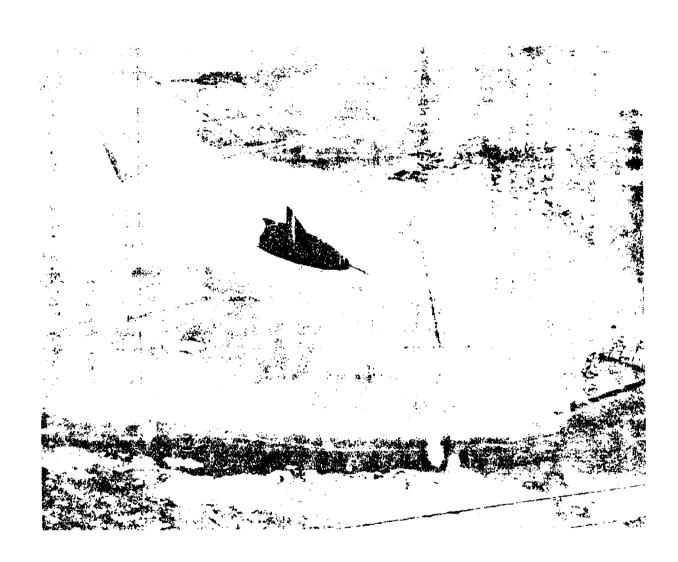


Figure 3 X-24A Wind Tunnel Data - S $_{2/3}$



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Final 5. AUTHOR(S) (First name, middle initial, last name)		 	
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This is Volume I of two volumes.	6510th Te Edwards A		fornia
of the X-24A lifting body from flight results with wind tunnel predictions Lift and drag characteristics were derations and flight conditions while Performance data were obtained over tranges of 0.26 to 1.5 and 1.5 to 19.0 cies were apparent between different flight test data generally exhibited than wind tunnel predictions with the tunnel L/D values were in reasonably number, vehicle control surface confiment on performance parameters were performance at low subsonic Mach number separation. In Volume II of this reparisons are made between the PRIME	t test dat were successive the vehice Mach numbe 6 degrees, wind tunn lower lifter result to good agree iguration, determined bers was a port, supe	ca and concessfully com onboated was interespection of the contest	rrelating these accomplished. Ind measured accel- in gliding flight. Igle of attack ively. Discrepan- ctions. However, ightly lower drag th test and wind Effects of Mach iding gear deploy- dation of flight d to tip fin flow erformance com-

were the same aerodynamic configuration (SV-5). A significant degradation in subsonic performance due to simulated ablative surface roughness was obtained in full scale wind tunnel tests. Flight testing of simulated ablatives on the X-24A was not pursued; however, the estimated

effects of the ablatives are discussed in Volume II.

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Security Classification	LIN	K A	LIN	кв	LIN	С	
KEY WORDS	ROLE	WT	ROLE	wT	POLE WT		
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lift drag glide flight powered flight ablatives							
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